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AIR FORCE SYSTEMS COMMAND

MANNED SPACECRAFT

by

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FOREIGN TECHNOLOGY DIVISION

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MANNED SPACECRAFT.

A. N. Ponomarev.

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The book discusses the Soviet spacecraft of the type "Vostok", "Voskhod", and about American ships "Mercury", "Gemini", and "Apollo", which are intended to be used for the delivery of man to the Moon; about future spacecraft for flight to other planets of solar system; orbital flight vehicles, intended for studying space, engines of space vehicles, are examined.

In book the problems, which appear with mastery/adoption of outer space, are briefly illuminated.

In description of specific samples and schematics of foreign flight vehicles, their equipment, which relate to the theme in question, there are used data published in the foreign press. The data published in the open Soviet press are used for the examination of Soviet samples of space technology.

Book is intended for the mass military reader who is interested in the development of manned space vehicles.

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U. S. BOARD ON GEOGRAPHIC NAMES TRANSLITERATION SYSTEM

Block	Italic	Transliteration	Block	Italic	Transliteration
А а	*А а*	A, a	Р р	*Р р*	R, r
Б б	*Б б*	B, b	С с	*С с*	S, s
В в	*В в*	V, v	Т т	*Т т*	T, t
Г г	*Г г*	G, g	У у	*У у*	U, u
Д д	*Д д*	D, d	Ф ф	*Ф ф*	F, f
Е е	*Е е*	Ye, ye; E, e*	Х х	*Х х*	Kh, kh
Ж ж	*Ж ж*	Zh, zh	Ц ц	*Ц ц*	Ts, ts
З з	*З з*	Z, z	Ч ч	*Ч ч*	Ch, ch
И и	*И и*	I, i	Ш ш	*Ш ш*	Sh, sh
Й й	*Й й*	Y, y	Щ щ	*Щ щ*	Shch, shch
К к	*К к*	K, k	Ъ ъ	*Ъ ъ*	"
Л л	*Л л*	L, l	Ы ы	*Ы ы*	Y, y
М м	*М м*	M, m	Ь ь	*Ь ь*	'
Н н	*Н н*	N, n	Э э	*Э э*	E, e
О о	*О о*	O, o	Ю ю	*Ю ю*	Yu, yu
П п	*П п*	P, p	Я я	*Я я*	Ya, ya

*ye initially, after vowels, and after ъ, ь; e elsewhere.
When written as ё in Russian, transliterate as yё or ё.

RUSSIAN AND ENGLISH TRIGONOMETRIC FUNCTIONS

Russian	English	Russian	English	Russian	English
sin	sin	sh	sinh	arc sh	$\sinh^{-1}$
cos	cos	ch	cosh	arc ch	$\cosh^{-1}$
tg	tan	th	tanh	arc th	$\tanh^{-1}$
ctg	cot	cth	coth	arc cth	$\coth^{-1}$
sec	sec	sch	sech	arc sch	sech^{-1}
cosec	csc	csch	csch	arc csch	csch^{-1}

Russian	English
rot	curl
lg	log

GRAPHICS DISCLAIMER

All figures, graphics, tables, equations, etc. merged into this translation were extracted from the best quality copy available.

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PREFACE.

The one who for the first time came up with the idea of being lifted into the air remains, apparently, unknown. The myths of distant antiquity speak about the manned flights on the eagles, the hawks, the winged horses. Ancient Greek myth narrates about Icarus, whom, desiring to leave the island, where he was in captivity, took off into the sky on wings of feathers, but, having flown too high, he approached the sun, and the wax which fastened the feathers, melted. Icarus perished.

How did humanity dream about put out into space. As long ago as 160 A.D. a fantastic flight to the Moon was described by the Greek satirist Lucian. The hero of his narrative with the aid of the wings of birds reached the Moon. Decided to be built up he, also, to the stars, but gods, after frightening the intrusion of man, took away in it wings.

Humanity emerged from infantile age, it increasingly more seriously related to its dream about flight into outer space. But many technical problems stood on the way to this. Overcoming the gravity force of the Earth was one of them.

K. E. Tsiolkovskiy wrote about the fact that "first go thought,

fantasy, fairy tale. After them marches scientific calculation".

Specifically, science fiction writers for the first time appeared with idea about possibility of overcoming gravity force of Earth. In 1865 there appeared the novel of the outstanding French writer, ancestor of the scientific fabulousness of Jules Verne "From the Earth to the Moon". In order to reach the Moon, its heroes created a gun, whose barrel had a length of 270 m and a bore of 2.7 m. The shot of 10-ton projectile with people in the direction of the Moon was produced from it.

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At the end of past century, when people began to dream about subjugation not only of Moon, but also such distant celestial bodies as planet Mars, in fantastic literature about interplanetary flights appeared new theme. In German writer Lassvitts' novel "On Two Planets" (1897), for the realization of flight to Mars it was proposed to have transfer station. As we see, this already agrees with with the contemporary projects of interplanetary flights.

Gradual storage by man of knowledge introduced corrections into utopian projects of interplanetary flight, making them with ever more real. Especially important in this respect was the discovery of the fact that the celestial bodies have material nature and according to their properties are similar to Earth.

Determination of true distances between celestial bodies and setting fact of three-dimensional/space limitedness of earth's atmosphere supplemented knowledge, without which it was not possible to work out technically substantiated, real designs of space vehicles.

As is known, as a basis of the contemporary concept about the structure of the universe lay great the discovery conducted by brilliant Polish scientific Nikolai Copernicus (1473-1543), who refuted views of Greek scientist Claudius Ptolemy (140 A.D.), which ruled until that time, who confirmed that Earth is the center of universe and all celestial bodies rotate around it. Copernicus advanced the hypothesis on which the sun is the central celestial body, and planets move around it along the enormous orbits. One of them - our Earth. It travels its path around the sun in 365.25 days.

Austrian astronomer Johann Kepler (1571-1630), an avid supporter of Copernicus's study, established that each planet moves along ellipse, in one of foci of which sun is located; that planets move in plane, passing through center of sun and finally that relation of squares of time of access of planets around sun is equal to relation of cubes of their medium solar distances.

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English naturalist and mathematician Isaac Newton (1643-1727) gave comprehensive proofs of reason for motion of planets and formulated law, which determines their motion. He revealed/detected

that two bodies mutually are attracted/tightened with the force, which depends, in the first place, on the mass of each of them and, in the second place, from the distance between them. This law, discovered by Newton, is called the law of universal gravity. The attraction of the Earth is developed so as if it proceeds from its center. Each body, which is located on the surface of the Earth, the aim is "to fall" as near as possible its center, but this it blocks the surface of the Earth. So there appears the force with which the body presses to its support. This force of pressure is called the weight of body, it is the consequence of the attraction of the Earth. In order to derive body from the gravitational field of the Earth, it is necessary to overcome the force of its attraction, for which to the body it is necessary to exert the force, which would act opposite to the line of force of the attraction of our planet. But if we impart the initial velocity of 7.91 km/s to a body, then at a certain orbital altitude it will not fall to the earth under the effect of the attracting force, but it will complete flight around it and thus it will be converted into the artificial Earth satellite. The flight of this satellite theoretically can continue infinitely. The complete revolution around the Earth will complete body in this case approximately in 1.5 hours.

During motion of the body in a circle, the centrifugal force, which acts in a direction opposite to the direction of the centripetal force, appears. In flight of body around the Earth with a velocity of 7.91 km/s the centrifugal force at all points of its orbit will be equal to centripetal force. Without taking into account the resisting

force to motion, it is possible to consider that the body will neither be removed fro. Earth nor approach it, i.e., it will fly in a circle.

Thus, speed of 7.91 km/s has very important value for motion characteristic of body in outer space. It is called a circular, orbital speed or orbital velocity in the field of gravity.

If body, directed to space, obtains speed, which exceeds 7.91 km/s[1], then it will move no longer in a circle but along an ellipse.

FOOTNOTE[1]. For the specific orbit altitude. ENDFOOTNOTE.

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The greater the speed imparted to the body, the more elongated there will be the ellipse. At the speed of 11.2 km/s the flight trajectory will take the form of a parabola and body, after overcoming the force of gravity, it will leave along this open curve into outer space. In connection with this the speed of 11.2 km/s is called parabolic, either speed of "release", or planet escape velocity.

At speed, which exceeds 11.2 km/s, trajectory will take form of hyperbola, which, as parabola, is open curve.

So-called mass inertia has an enormous effect on all forms of motion in nature. With a change in the speed of motion the body exerts the known resistance, for overcoming which to it it is

necessary to make effort. This change in the speed of motion into the specific time interval is called acceleration. For the unit acceleration accept a change in the speed of the freely falling/incident body on the surface of the Earth in 1 s. It equal to 9.81 m/s^2 is designated by letter g.

Acceleration is accepted to characterize by amount of g-force, which shows, by how many times actual acceleration is more than acceleration of gravity. It is obvious that the greater the acceleration, the greater the g-force.

The g-force compulsorily appears during expulsion into outer space. The acceleration continuously increases with the start of spacecraft during rocket thrust-chamber firing. Let us assume that the acceleration achieved 39.24 m/s^2, i.e., rocket obtained acceleration 4 g (9.81×4). Consequently, g-force is also equal to four units. But this means that the "weight" of each crew member of the ship will increase 4 times. The word "weight" we included in quotation marks, since it makes relative sense. Here it would be more correct to say thus: gravity, which in this case acts on the man, will increase 4 times.

Acceleration, and consequently, g-force can reach considerably larger values. It is clear that to the body of man of weight in this case terrible gravity.

Being located in the vertical position, man could not remove it.

Value of g-force transferred by man and time of its action usually are found in inverse dependence: the less time of action of g-force, the more it can be in value. Consequently, it is not possible to indicate some specific maximum g-force, which man can maintain/withstand. Everything depends on the time of its action. Experiments proved that man, being located in the vertical position, transfers sufficiently well g-forces to 8 g for the time of 3 s and to 5 g for the time of 12-15 s. During the momentary effect (less than 0.1 s) of men it transfers twenty-fold g-forces.

From aforesaid it follows that task of designers of manned space vehicles is creation of such flight conditions, under which g-forces would be safe for man.

One should recall that during flights on contemporary jet aircraft high accelerations also appear under some conditions and respectively - heavy overloads. In essence these are centrifugal accelerations in flight along the curve. It is possible to obtain them, also, in the ground-based laboratory with the aid of the so-called centrifuge. Experiments on the centrifuge and test flights made possible to establish that man was capable to transfer the heavy overloads, if they were directed perpendicular to the longitudinal axis of his body, in other words, if man lies/rests, since in this

position the blood is not accumulated/stored in some part of the body. Some people during experiments transferred in this position of the g-force to 10 g during 30 s. In this case in them there was not observed any disorders.

In resolution of problems of flight into space Soviet inventors and scientists played a salient role. It is possible to call them rightfully the pioneers of cosmonautics.

With question of flights into outer space dealt Russian inventor revolutionary N. I. Kibal'chich.

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In the secret tsarist archives were discovered the documents, which contain the information about the fact that Kibal'chich in the casemate of Petropavlovsk stability, where it was included in 1881 for the attempt on the tsar, worked on the design of the flight vehicle, which moves with the aid of the solid-propellent rockets. It confirmed that this engine is the means of reaching the maximum speeds and overcoming the attracting force of the Earth.

Especially there should be mentioned the name of Constantine Eduardovich Tsiolkovskiy. Its work "Investigation of outer space by reaction instruments" (1903) played the salient role in the development of the theory of reactive motion and in the substantiation of the possibility of applying jet-propulsion technology for the

remote space flights. K. E. Tsiolkovskiy proposed a schematic of liquid propellant rocket engine, which in the principle is used at present.

After the great October Socialist Revolution K. E. Tsiolkovskiy continued fruitfully to work on questions of mastery of outer space with wide support to Communist Party. He became the teacher of the pleiad of scientists, who dedicated their life space flights.

On the theory of the motion of bodies of variable mass, which include rocket, worked also I. V. Meshcherskiy. In his works, published into 1897 and 1904, he gave the fundamental equations of the dynamics of particle of variable mass.

High value for the development of rocket engineering is the work of F. A. Tsander, Yu. V. Kondratyuk, and other scientists, published in 1924-1959. Academician S. P. Korolev played the salient role in the development of rocket and space technology (1906-1966).

Since beginning of XX century and to the present are published many works on problems of space flights, is organized not one society, which was being occupied by these questions. It is possible, in particular, to mention about the organization in Moscow in 1924. Central bureau for the investigation of rocket problems. Into the leading center of this bureau entered K. E. Tsiolkovskiy, N. Ye. Rynin and many others.

To this author, at that time had only begun service in aviation. it was difficultly assume, listening to professor Rynin, a scientist of wide profile, that only after 40 years there will appear supersonic aircraft and manned spacecraft.

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In 1924 in Moscow the Society for the study of inteplanetary flights was also organized. In 1927 it arranged the first international exhibition dedicated to inteplanetary flights.

In other countries much attention was also given to problems of space flights and the creation of jet engines, but one should speak especially about this. Here we will only point out that the extensive work directed at the creation of the rocket as weapons, as the means of destruction, was conducted in the period of the Hitler mode in Germany. This rocket on 3 September, 1944, was sent from the German range to England.

The creation in the Soviet Union of powerful ballistic missiles and their successful launching in 1957 were bright evidence of outstanding achievements of Soviet rocket engineering. On 4 October, 1957, in the USSR there was launched the first in the world artificial Earth satellite. Its radio signals notified peace/world about the beginning of the space age. Soviet people first put into practice of the idea of their great compatriot of K. E. Tsiolkovskiy, they left

into outer space. Entire progressive humanity was enraptured by this great victory of Soviet people, which demonstrated the bright rates of the scientific-technical progress in the USSR.

By the way, the Western World at that time did not doubt that the first to launch a satellite would be the USA. American radio, the press and television, without spairing the colors, in every way painted the forthcoming launch of satellite "Vanguard" with a weight only of 1.47 kgf.

Then it is still difficult it was difficult to assume that less than four years after launch of first satellite will be launched into outer space apparatus with man aboard. Indeed for this it was necessary to create the heavy spacecraft, whose equipment would ensure the normal stay of man under the space conditions and weightlessness; to solve the problems not only injection of ship into orbit, but also his returns to the Earth and safe touchdown in the prescribed/assigned area; to ensure the complete reliability of the work of the communication systems with the Earth for the control/check of the state of the cosmonaut. All this presented enormous difficulties, and were they permitted within the shortest period.

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On 12 April 1961, in the Soviet Union there was put into near earth orbit the first in the world spacecraft with man aboard. This spaceship was called "Vostok", its Soviet airman-cosmonaut Yuri

Gagarin piloted. Minimum removal/distance from the surface of the Earth was 175 km, maximum - 302 km. Spacecraft weighed 4725 kgf. In 108 min Yuri Gagarin flew around terrestial globe. Then the retro-engine was switched on and the ship began to descend for the touchdown. After 10 hours and 55 min Moscow time the spaceship "Vostok" satisfactorily completed landing in the prescribed/assigned area of the Soviet Union. The obelisk is now on the spot of touchdown erected in the sign of this flight.

To all are known words of Konstantin Eduardovich Tsiolkovskiy: "Earth is the cradle of reason, but one cannot always live in a cradle". Time came, and man left the cradle, there was carried out flight into space and it returned to the earth. Path to the planets is opened.

Academician S. P. Korolev and the first cosmonaut in the world, Yu. A. Gagarin.

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Subsequently, Soviet space satellite vehicles began to complete ever more and more endurance flights. In less than four months into space there was launched the satellite vehicle "Vostok-2", piloted by G. Titov. This flight continued more than days.

Group multiday flight of cosmonauts A. Nikolayev and P. Popovich became new enormous reaching/achievement of Soviet science.

The sixth year of space age was marked by a joint multiday flight of cosmonaut V. Bykovskiy and first in the world woman-cosmonaut V.

Tereshkova. This flight finally demonstrated the possibility of the prolonged stay of people under the conditions for outer space and retaining/maintaining in this case the efficiency.

On 12 October 1964, was begun new stage in mastery/adoption of outer space. It was created and completed its first voyage into the celestial they gave space multiplace space ship "Voskhod". For the first time in the history of humanity space the friendly collective of Soviet people visited. Aboard the ship "Voskhod-1" the cosmonauts - craft commander V. Komanrov, scientific worker K. Feoktistov and doctor B. Yegorov sixteen times flew around terrestial globe. During the flight the spacecraft crew conducted great scientific work in the field of cosmonautics, biology, medicine and other sciences.

Space ship "Voskhod "was injected into orbit by more powerful carrier rocket than "Vostok". The cabin/compartment of space ship "Voskhod" made it possible for cosmonauts to complete flight in the usual clothing, without the pressure suits, which trouble their actions. Furthermore, aboard the ship there were installed the more advanced communication systems with the Earth, intravehicular communication was realized.

In flights aboard "Vostok" spacecraft and "Voskhod-1" Soviet cosmonauts conducted large scientific observations and experiments, rated/estimated work of installed equipment, checked work of means of connection/communication of control and orientation. All this made

possible to our scientists and to designers repeatedly to test and to master different systems of spacecraft, to study efficiency of man under the conditions of weightlessness. The acquired knowledge and experience was used for the resolution of the problem of the output of man from the ship into space.

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Our scientists and designers created two-place/dyadic manned spacecraft "Voskhod-2", whose design made it possible to carry out the putting of man into outer space.

On 18 March 1965, into space was launched the ship "Voskhod-2" with cosmonauts Pavel Belyayev and Aleksey Leonov.

On the second orbit A. Leonov egressed into outer space in a special pressure suit, he was removed from ship at 5.35 m, successfully were conducted outlined research and it returned to ship. In outer space the cosmonaut stayed more than 20 min, including in open space, 12 min. The egress of A. Leonov from the ship and his subsequent return to the ship were accomplished by the method of locking. On leaving the ship and in entire period of determination in outer space he breathed oxygen, which enters from the tanks/balloons of the autonomous power-supply system of pressure suit.

Pressure suits of cosmonauts had multilayer sealed/pressurized shell, which allows on leaving into outer space to maintain excess

pressure, necessary for guaranteeing normal vital activity, within them. In the pressure suit the ventilation systems and oxygen feed/supply were provided for also.

Manual control for landing was for the first time used aboard this ship.

Soviet scientists continuously continue space research. To the coma of the starting/launching of the manned spacecraft, for this purpose are utilized artificial Earth satellites and automatic interplanetary space stations. On 16 March, 1962, in orbit around the Earth was launched artificial satellite "Kosmos-1". Since then it passed 6 years. And thus already more than 200 satellites is launched into space. With their aid the scientists obtained the large volume of scientific information on all questions, which concern flight into space, namely - the new information about the terrestrial ionosphere and the conditions for radiowave propagation; about distribution and intensity of the charged particles; given, connected with the improvement of the elements of the design of space vehicles. etc.

With the aid of artificial communication satellites "Molniya-1" there are accomplished telecasts to large distances.

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Soviet cosmonauts (from left to right): the first row - V. M. Komarov [deceased], Yu. A. Gagarin [deceased], V. V. Nikolayeva-Tereshkova, A. G. Nikolayev, K. P. Feoktistov, P. I. Belyayev; the second row - A. A. Leonov, G. S. Titov, V. F. Bykovskiy, B. B. Yegorov, P. R. Popovich.

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Continuous improvement of the rockets made it possible to launch into space whole scientific laboratories. With the aid of such laboratories as, for example, "Proton" and "Elektron", scientists conducted a number of very important scientific investigations in space, in particular, studied space particles of high and superhigh energies.

For purposes of adjustment of apparatuses, systems and equipment,

suitable for long-distance flights into space, in 1961-1967 were produced the launching of scientific stations to Venus, Mars, Moon. The first scientific investigations of circumlunar space produced "Luna-1" station, launched in 1959. Since then were launched 14 automatic stations, which were utilized for studying of circumlunar space and physical properties of the Moon.

In 1966 to Moon is for the first time in history produced soft landing "Luna-9" station. At the end of 1966 the "Luna-13" station reached the Moon and softly "landed on the moon".

Salient scientific reaching/achievement of Soviet science and technology was realization on 18 October, 1967, landing on Venus automatic station "Venera-4". The flight of station to Venus it continued 4 months, within this time it was removed from the Earth up to the distance about 80 million kilometers. Communication was accomplished during entire flight time with the station. The command for its descent was given during the approach of station to Venus from the Earth. Station transmitted the data of large scientific value about the atmospheric parameters of Venus.

At the end of October of 1967 new news about the unprecedented scientific experiment, carried out in our country, went around the entire world. Was realized the first in the world automatic mating and conjointing of artificial Earth satellites "Kosmos-186" and "Kosmos-188" in orbit of the Earth. The flight of the butted complex

continued for 3 hours, 30 min. During April of 1968 the experiment on the automatic mating in space, which confirmed the structural/design reliability of the developed systems, was repeatedly carried out.

Outer space is studied, as the reader already knows, with the aid of artificial Earth satellites, scientific automatic stations and apparatuses with man aboard.

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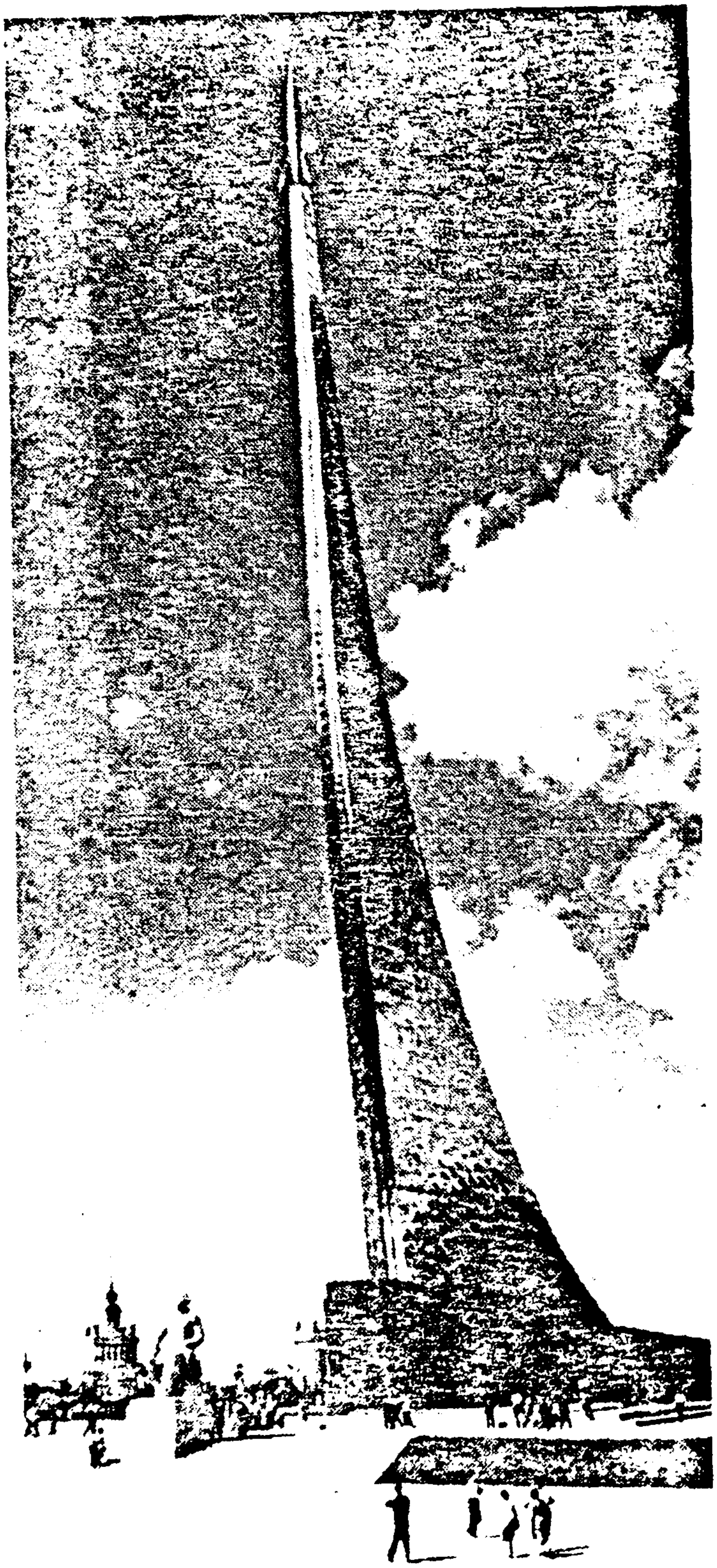

Monument to subjugators of space in Moscow.

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The creation of space scientific stations of the type "Proton" and "Elektron" for the investigation of near-earth outer space, the realization of the a number of flights of automatic stations to the Moon, landing on the surface of Venus, the launching of a large quantity of satellites of a series "Kosmos" with different scientific purposes and the series/row of other bright successes in the mastery/adoption of outer space characterizes capacity and multiplaned character of Soviet space program. The author did not set for himself as a goal to illuminate all achievements of Soviet and foreign science and technology in the field of cosmonautics, since the volume of this book does not make it possible to do this. Therefore he was restricted to the task of introducing the reader only to one of the most important means of the study of space - by the manned space vehicles. Of course the author does not pretend to the complete illumination of all questions concerning the theme in question. The purpose of the book is only to introduce briefly to the reader these apparatuses, and to also briefly present all connected with them questions.

In conclusion the author considers it his debt to thank doctor of technical sciences, V. F. Pavlenko, for his attentive review of the manuscript.

Author.

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Chapter 1.

MECHANICS OF SPACE FLIGHTS.

The concept "space" is sufficiently wide; speaking about space, we usually have in mind the entire universe. However, flights in limits of the solar system at present in essence are examined, or more precisely - in the zone of the orbits of the Earth, Venus and Mars.

The basic force which determines the motion of an artificial satellite or spaceship in outer space is the gravity force of planet, around which orbits this satellite (ship), or the gravity force of sun (in flight in outer space out of gravitational field of planet).

Flight trajectory of artificial Earth satellite or spacecraft by earth's orbit is determined in by ground field of gravity of Earth. Therefore, with a sufficient precision/accuracy it is possible to find it, if we disregard the effect of the gravitational poles of the Sun and Moon. To consider their attracting force is necessary only for obtaining the results of very high precision. During the interplanetary flight of spacecraft the gravitational field of the Earth will have an essential effect on it only in the initial period, when it will be located in immediate proximity to the Earth. The attraction of the Earth can be already disregarded in a certain stage of the flight of ship, since the fundamental effect on the flight in

this case will be exerted by the gravitational pole of the Sun, and from a specific distance - and the planet toward which is bound the spacecraft.

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Table 1. Basic trajectory and physical data of the sun and planets.

(1) Наименование	(2) Солнце	(3) Планеты солнечной системы								
		(4) Меркурий	(5) Венера	(6) Земля	(7) Марс	(8) Юпитер	(9) Сатурн	(10) Уран	(11) Нептун	(12) Плутон
Масса, кг (13)	$1.99 \cdot 10^{30}$	$3.17 \cdot 10^{23}$	$4.87 \cdot 10^{24}$	$5.98 \cdot 10^{24}$	$6.40 \cdot 10^{23}$	$1.90 \cdot 10^{27}$	$5.69 \cdot 10^{26}$	$8.70 \cdot 10^{25}$	$1.03 \cdot 10^{26}$	$5.40 \cdot 10^{24}$
Радиус, км (14)	$6.96 \cdot 10^{5}$	$2.42 \cdot 10^{3}$	$6.20 \cdot 10^{3}$	$6.38 \cdot 10^{3}$	$3.40 \cdot 10^{3}$	$7.14 \cdot 10^{4}$	$6.04 \cdot 10^{4}$	$2.38 \cdot 10^{4}$	$2.21 \cdot 10^{4}$	$7.20 \cdot 10^{3}$
Относительное ускорение силы тяжести (g_i/g) (15)	28.039	0,386	0.895	1.000	0.381	2.918	1.326	0.961	1.000	0.492
Наклонение экватора к орбите, град (16)	—	7°	—	23°30'	24°	3°	26°45'	98°	151°	—
Период вращения, сут. час. мин. сек (17)	25, 9, 7, 12	87, 23, 15—	—	—23, 56, 4	—24, 37, 23	—9, 55, 41	—10, 14, 24	—10, 8—	—15, 40—	—
Удельная потенциальная энергия на поверхности, дж/кг (18)	$1.91 \cdot 10^{11}$	$8.74 \cdot 10^{6}$	$5.24 \cdot 10^{7}$	$6.25 \cdot 10^{7}$	$1.25 \cdot 10^{7}$	$17.7 \cdot 10^{8}$	$6.28 \cdot 10^{8}$	$2.44 \cdot 10^{8}$	$3.08 \cdot 10^{8}$	$6.00 \cdot 10^{7}$
Скорость для выхода из сферы притяжения V_e, км/сек (19)	618	4.18	10.2	11.2	5.00	59.7	35.4	22.0	24.8	10.0
Среднее расстояние от Солнца, км (20)	—	$5.79 \cdot 10^{7}$	$1.081 \cdot 10^{8}$	$1.495 \cdot 10^{8}$	$2.273 \cdot 10^{8}$	$7.778 \cdot 10^{8}$	$1.426 \cdot 10^{9}$	$2.869 \cdot 10^{9}$	$4.495 \cdot 10^{9}$	$5.899 \cdot 10^{9}$
Радиус орбиты минимальный, км (21)	—	$4.59 \cdot 10^{7}$	$1.071 \cdot 10^{8}$	$1.467 \cdot 10^{8}$	$2.060 \cdot 10^{8}$	$7.400 \cdot 10^{8}$	$1.344 \cdot 10^{9}$	$2.728 \cdot 10^{9}$	$4.450 \cdot 10^{9}$	$4.430 \cdot 10^{9}$
Радиус орбиты максимальный, км (22)	—	$6.97 \cdot 10^{7}$	$1.088 \cdot 10^{8}$	$1.520 \cdot 10^{8}$	$2.483 \cdot 10^{8}$	$8.150 \cdot 10^{8}$	$1.503 \cdot 10^{9}$	$3.000 \cdot 10^{9}$	$4.530 \cdot 10^{9}$	$7.410 \cdot 10^{9}$
Сидерический период обращения (земной год = 1) (23)	—	0.241	0.615	1.000	1.881	11.862	29.458	84.015	164.788	247.697
Наклонение орбиты к эклиптике (24)	—	7°0'14.3"	3°23'39.2"	0°0'0"	1°50'59.8"	1°18'19.5"	2°29'23.9"	0°46'23"	1°46'25.8"	17°8'38.4"
Средняя скорость движения по орбите, км/сек (25)	—	47.86	35.01	29.76	24.11	13.06	9.64	6.78	5.47	4.84
Температура поверхностного или облачного слоя, °C (26)	5527°	337°	—39°	14°	—20°	—143°	—145°	—170°	—210°	—230°
Альбедо, A (27)	—	0.069	0.59	0.39	0.15	0.44	0.42	0.45	0.52	0.16

Key: (1). Designation. (2). Sun. (3). Planets of solar system. (4). Mercury. (5). Venus. (6). Earth. (7). Mars. (8). Jupiter. (9). Saturn. (10). Uranus. (11). Neptune. (12). Pluto. (13). Mass, kg. (14). Radius, km. (15). Relative acceleration of gravity .... (16). Inclination of equator to orbit, deg. (17). Period of rotation, day, hour, min, s. (18). Specific potential energy on surface, J/kg. (19). Speed for output from sphere of attraction ... km/s. (20). Average distance from sun, km. (21). Radius of orbit, minimum, km. (22). Radius of orbit, maximum, km. (23). Sidereal period of orbit (terrestrial year=1). (24). Orbit inclination to ecliptic. (25). Average speed of orbital movement, km/s. (26). Temperature of surface or cloud layer, °C. (27). Albedo, A.

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Table 2. Basic Trajectory and physical data of the greatest satellites of planets.

(1) Наименование	(2) Земля	(3) Марс		(4) Юпитер				(5) Сатурн			(6) Уран	(7) Нептун
	(8) Луна	(9) Фобос	(10) Деймос	(11) Ио	(12) Европа	(13) Ганимед	(14) Каллисто	(15) Диона	(16) Рея	(17) Титан	(18) Титания	(19) Тритон
Масса, кг . . . (20)	$7{,}34 \cdot 10^{22}$	$6{,}4 \cdot 10^{16}$	$6{,}4 \cdot 10^{14}$	$8{,}54 \cdot 10^{22}$	$4{,}82 \cdot 10^{22}$	$1{,}52 \cdot 10^{23}$	$8{,}55 \cdot 10^{22}$	$1{,}08 \cdot 10^{21}$	$2{,}28 \cdot 10^{21}$	$1{,}21 \cdot 10^{23}$	—	$1{,}56 \cdot 10^{22}$
Радиус, км . . . (21)	$1{,}738 \cdot 10^{3}$	8,000	4,000	$1{,}775 \cdot 10^{3}$	$1{,}550 \cdot 10^{3}$	$2{,}800 \cdot 10^{3}$	$2{,}525 \cdot 10^{3}$	$7{,}00 \cdot 10^{2}$	$9{,}25 \cdot 10^{2}$	$2{,}475 \cdot 10^{3}$	$8{,}03 \cdot 10^{2}$	$2{,}40 \cdot 10^{3}$
Относительное ускорение силы тяжести (22)	0,165	0,0007	0,0003	0,185	0,137	0,132	0,091	0,015	0,018	6,134	—	0,022
Удельная потенциальная энергия на поверхности, дж/кг . (23)	$2{,}81 \cdot 10^{6}$	53,4	10,7	$3{,}21 \cdot 10^{6}$	$2{,}07 \cdot 10^{6}$	$3{,}62 \cdot 10^{6}$	$2{,}26 \cdot 10^{6}$	$1{,}03 \cdot 10^{5}$	$1{,}64 \cdot 10^{5}$	$3{,}28 \cdot 10^{6}$	—	$5{,}17 \cdot 10^{6}$
Скорость для выхода из сферы притяжения V_e, км/сек . (24)	2,37	0,0103	0,0046	2,53	2,04	2,69	2,12	0,45	0,57	2,55	—	1,02
Среднее расстояние от планеты, км . (25)	$3{,}844 \cdot 10^{5}$	$9{,}4 \cdot 10^{3}$	$2{,}35 \cdot 10^{4}$	$4{,}23 \cdot 10^{5}$	$6{,}714 \cdot 10^{5}$	$1{,}071 \cdot 10^{6}$	$1{,}884 \cdot 10^{6}$	$3{,}775 \cdot 10^{5}$	$5{,}272 \cdot 10^{5}$	$1{,}222 \cdot 10^{6}$	$4{,}392 \cdot 10^{5}$	$3{,}537 \cdot 10^{5}$
Эксцентриситет (26)	0,0549	0,0190	0,0031	0	0,0003	0,0015	0,0075	0,0020	0,0009	0,0289	0,0023	0,000
Сидерический период обращения (земные сутки = 1) (27)	27,322	0,319	1,262	1,769	3,551	7,154	16,689	2,737	4,518	15,945	8,706	5,877

Key: (1). Designation. (2). Earth. (3). Mars. (4). Jupiter. (5). Saturn. (6). Uranus. (7). Neptune. (8). Moon. (9). Phoebe. (10). Deimos. (11). Io. (12). Europa. (13). Ganymede. (14). Callisto. (15). Dione. (16). Rhea. (17). Titan. (18). Titania. (19). Triton. (20). Mass, kg. (21). Radius, km. (22). Relative acceleration of gravity. (23). Specific potential energy on surface, J/kg. (24). Speed for output from sphere of attraction ..., km/s. (25). Average distance from planet, km. (26). Eccentricity. (27). Sidereal period of orbit (terrestrial days=1).

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Thus, depending on what kind of gravitational field has the

predominant effect on spacecraft, flights into space conditionally can be divided into flights around this planet, when acting forces of other planets and sun are low and it is possible not to take them into consideration, and to flights between planets, when it is not possible to disregard action of gravitational field of sun, but in individual sections also of planets. The first flights are called geocentric (around the Earth), selenocentric (around the Moon) and so forth, the second - interplanetary.

Laws of motion of satellites and spacecraft (without applicable thrust) just as laws of motion of any celestial bodies, are established/installed by celestial mechanics. As its basis lies the law of universal gravitation of Newton.

For determining the character of motion of flight vehicles in outer space, first of all, it is necessary to know characteristic features of space flight.

SPACE FLIGHT CONDITIONS.

To the space flight condition first of all relate: gravitational poles of the Sun and planets, environmental parameters and meteoritic flows.

Gravitational fields. The forces of gravitational fields are basic external forces which act on flight vehicles in the space

flight. These forces are determined by the values of the masses of planets, by distance to the center of gravitational mass and value of universal gravitational constant. Tables 1 and 2 give the values of masses for the celestial bodies of the solar system, and also their basic trajectory and physical data [2].

Motion of space vehicle in gravitational field in the absence of other forces (thrust force and resistance) is characterized by fact that the total energy remains constant:

$$E = \frac{G}{g}\left(\frac{V^2}{2} - \Sigma f \frac{M_i}{R_i}\right) = \text{const}, \qquad (1)$$

where f - universal gravitational constant;

M_i - value of mass of gravitational center;
R_i - distance to center of gravitational mass.

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During motion in gravitational field of prevailing action of one celestial body (Suns, planet or its satellite), when effect of other celestial bodies can be disregarded/neglected, equation (1) is simplified and is written/recorded in this form:

$$E = \frac{G}{g}\left(\frac{V^2}{2} - f \frac{M}{R}\right) = \text{const}. \qquad (2)$$

Specific potential energy

$E_n = f\frac{M_i}{R_i}$ on surface of celestial body characterizes power

expenditures, necessary for breakaway of the space vehicle from sphere of its attraction. The same expenditures can be characterized with velocity, necessary for the breakaway from the sphere of the attraction

$V_e = \sqrt{2f\frac{M}{R}}$. The values of specific potential energies and speeds V_e are given in Tables 1 and 2.

Environmental parameters. For interplanetary space flight characteristically substantial change along the flight trajectory of pressure, temperature of environment, density. A change in these parameters has important value with the flights in immediate proximity to the planet of start and the destination planet, in transit through their atmosphere. Fig. 1 gives vertical distribution of pressure p_H, temperature T_H, density ρ_H in the atmosphere of the Earth. As can be seen from figure, pressure and density decrease with an increase in altitude. Pressure and density in the atmosphere of other planets also fall from an increase in altitude.

In interplanetary state space of gaseous medium is determined mainly by corpuscular radiation of sun (by flow of ionized atoms of hydrogen - protons). The intensity of the flow of protons is inversely proportional to the square of distance from the sun. At the same time in the interplanetary space there is a powerful/thick energy flux, which is the electromagnetic radiation of the sun with the continuous spectrum. Rate of flow of light solar radiation also is

inversely proportional to the square of distance from the sun.

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The interaction between luminous radiation and body reflecting (or absorbing) light causes pressure not body. The value of light pressure depends on the radiated power; in orbit of the Earth its maximum value composes $\sim 0{,}928 \cdot 10^{-6}$ kgf/m² [2].

In flight near celestial bodies to space vehicle they can exert noticeable effect and energy fluxes from intrinsic emission of these bodies, and also reflected from them solar radiations. The solar radiations reflected from the surface of celestial bodies it is accepted to characterize by the magnitude of the albedo of the surface of this body - A. The Albedos is the ratio of luminous flux to the initial reflected from the celestial body. The magnitudes of the albedo for the planets of the solar system are given in Table 1.

Total effect of energy fluxes and internal energy of space vehicle makes it possible to determine its thermal condition in flight.

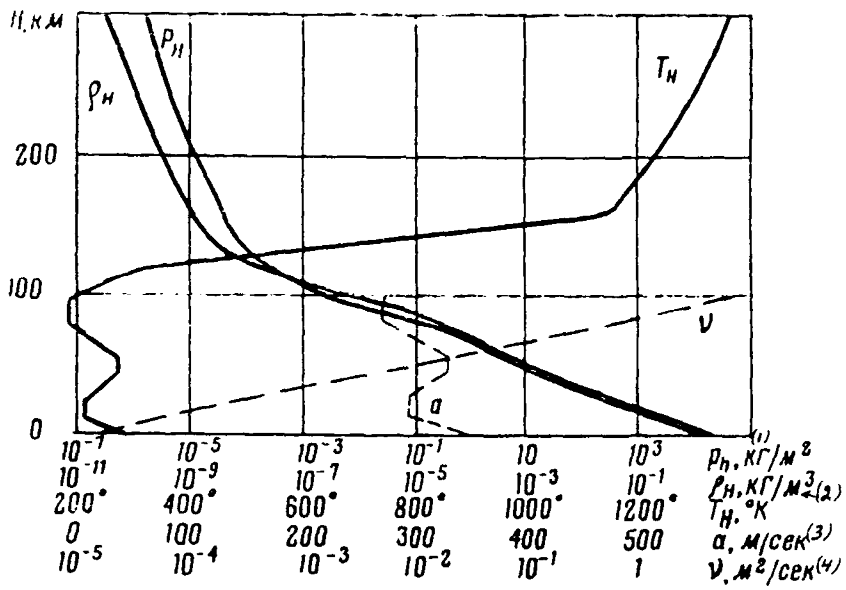

Fig. 1. Change in pressure, density and temperature with altitude in the atmosphere of Earth; broken lines show the speed of sound a and kinematic viscosity ν.

Key: (1). kg/m². (2). kg/m³. (3). m/s. (4). m²/s.

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Besides energy fluxes, a considerable effect on space flight can be rendered by radiation fluxes in space. They adversely affect the organism of a human and animals. Therefore from their effect, especially during the endurance flights, is necessary protection. As a result of the interaction of cosmic rays with the magnetic field of the Earth the radiation belts with the high intensity of the charged particles are formed. Flight in these flanges/belts presents considerable biological danger.

Meteoritic flows. The character of distribution and the intensity of meteoritic flows in the interplanetary space of the solar system cannot but affect the selection of the rational flight trajectories of space vehicles. Furthermore, it is necessary to provide for the protection of space vehicles from the possible rendezvous with the meteoritic flows.

The average speed of meteoritic particles near Earth vary within the range of 15 to 28 km/s. The greater speeds correspond to the greater masses of particles. The masses of particles vary in the limits from 10^{-12} up to 1 g.

TRAJECTORY ELEMENTS AND STAGES OF MOTION OF SPACE VEHICLES.

Flight trajectory of space vehicle (ballistic missile, artificial Earth satellite, spacecraft, etc.) consists of the following sections: active, passive and entry into the atmosphere (Fig. 2).

Powered phase (1-2). On it to flight vehicle is communicated the required for further flight speed. The speed of its motion in the dense layers of the atmosphere is comparatively small during the aircraft launching from the Earth. However, at the end of the powered phase, when the speed of motion is led to the near-space, flight is accomplished/realized in the upper air.

Powered phase of flight vehicle, intended for flight to Moon or

to Mars, to Venus and to other planets, as it is divided into two parts.

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Its injection into the orbit around the Earth is first produced, and then the engines, which impart to apparatus the second cosmic velocity, are switched on, and it passes in orbit of flight to the celestial body.

Free-flight phase (2-3). On the reaching/achievement by the space vehicle of given speed the work of engines ceases. Powered flight on this concludes. Further flight is accomplished/realized under the action of the gravitational field of this planet, its satellite or sun. Furthermore, on the space vehicle acts the aerodynamic resisting force of the atmosphere of planet. Although the amount of this force is insignificant, during the endurance flight it can exert essential influence on the flight speed and form of trajectory.

Section of entry into the atmosphere (3-4). For the return to the earth after flight into space flight vehicle is braked and enters into the dense layers of the atmosphere. Braking is accomplished/realized by an engine installation, if these are artificial satellite or spacecraft. Space vehicles enter into the dense layers of the atmosphere with the near-space speeds. In this case heat fluxes and aerodynamic loadings sharply grow/rise.

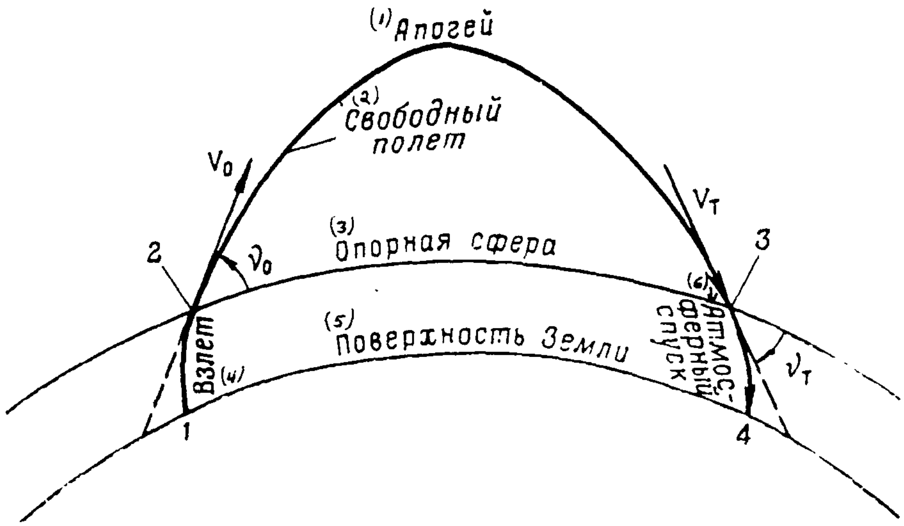

Fig. 2. Flight trajectory of ballistic missile.

Key: (1). Apogee. (2). Free flight. (3). Reference sphere. (4). Takeoff. (5). Surface of Earth. (6). Atmospheric descent.

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Motion of space vehicle in the powered phase.

Powered flight trajectory, in which work engines, is intended for communication/report to flight vehicle of desired values of speed, altitude and flight path angle to the horizon. The character of the motion of space vehicle in this section is determined by the program of motion and by the method of start. There can be not only ground-based, but also air starts.

Based on the example of the removal of artificial satellite in orbit of Earth let us examine character of its motion in powered

flight trajectory.

In order to put an artificial satellite into the earth's orbit, the carrier rocket must supply it to the required altitude and accelerate to a speed at which it would continue flight on closed curve around Earth (to orbit). The simplest orbit of artificial Earth satellite is the circular orbit, i.e., such in which the satellite completes motion with the constant velocity at constant distance from the center of the Earth. But are there such speeds to which it is necessary to accelerate satellite so that it would accomplish/realize the flight along the circular orbit?

It is known that for central gravitational gravitational field, according to law of universal gravitation, is determined by expression

$$G_r = mg_r = f\frac{Mm}{r^2} = \frac{Km}{r^2}, \qquad (3)$$

where f - universal gravitational constant; its value is identical for all bodies;

$$f = \frac{1}{3866^2}\,\frac{{}^{(1)}\text{см}^3}{{}^{(2)}\text{сек}^2};$$

Key: (1). cm^3. (2). gs^2.

K=fM - gravitational parameter; for gravitational field of Earth $K=398620\ km^3/s^2$;

M - mass of planet;

m - mass of flight vehicle;

r - distance from center of Earth to flight vehicle;

g_r - acceleration of gravity at a distance r from the center of Earth.

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If the ship is found in a circular orbit, then the attracting force proves to be to equal the centrifugal force with uniform motion at a rate of V in circle radius r, i.e.

$$\frac{mV_{\text{кр}}^2}{r} = \frac{Km}{r^2}.$$

Hence

$$V_{\text{кр}} = \sqrt{\frac{K}{r}}.$$

It is accepted to call, as has already been indicated, this speed circular, orbital or orbital velocity.

It is easy to show that for a circular orbit with an altitude above the surface of Earth 100 km an orbital speed of 7.85 km/s, and at altitude of 300 km - 7.73 km/s. This, of course, does not mean that to derive satellite on the altitude of 300 km it is simpler than to the altitude of 100 km. The fact is that with an increase in altitude of removal the speed losses to overcoming of the gravitational field of the Earth grow, the increase in these losses with an increase in altitude prevailing over the decrease of orbital speed.

It is necessary to consider in examination of active track-out phase of artificial Earth satellite that removal is accomplished/realized from rotating Earth. Therefore the absolute final velocity of satellite will be composed of its speed with respect to the fixed Earth (relative speed) and velocity of following of the daily rotation of the Earth. For the constant absolute final velocity the relative velocity of satellite (carrier rocket) can be less or more than the absolute final velocity in the dependence on the orbit inclination, i.e., depending on the azimuth of injection point. The less the orbit inclination, the less the required relative velocity, i.e., the more we utilize rotation of the Earth for the communication/report of the necessary velocity to satellite. Thus, the maximum velocity, which can be utilized due to the rotation of the Earth during the zero inclination (starting/launching it is accomplished/realized eastwards in the equatorial plane), it is 465 m/s.

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During the large inclinations (more than 90°) it would be necessary even to overcome the peripheral speed of the rotation of the Earth. This means that the relative velocity must be more than absolute.

With injecting satellite into orbit program of change in flight path the angle to the horizon has a high value. In the initial phase of flight the trajectory of carrier rocket is analogous to the

trajectory of ballistic missile. It is the vertical launching phase and a comparatively steep initial section. This character of trajectory provides the greatest simplicity of start and maximally rapid output of satellite from the dense layers of the atmosphere even before reaching/achievement by it of high flight velocities. Further control system provides predetermined program of the rotation of rocket so that the normal forces continuously bend trajectory before reaching/achievement of the the assigned flight path angle to the local horizon.

In simplest case for orbits with relatively small perigee altitude program of turn of rocket must ensure zero angle of inclination/slope of velocity vector to local horizon up to moment of achieving to given speed and altitude (Fig. 3a).

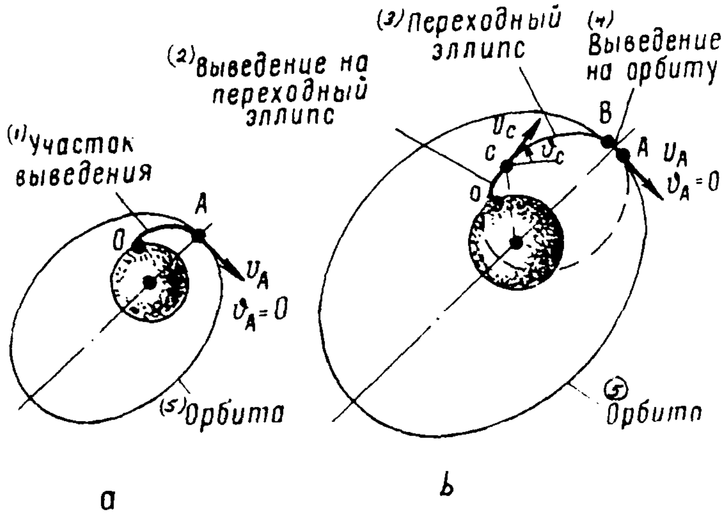

Fig. 3. Different methods of injecting a satellite into orbit: a) with low altitude of perigee; b) with high perigee altitude.

Key: (1). Track-out phase. (2). Injection to transfer ellipse. (3). Transfer ellipse. (4). Orbital injection. (5). Orbit.

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For obtaining the orbits with the largest possible perigee altitude the program of removal must be connected with a change of the mass of rocket as a result of the fuel consumption with the work of engines and a change in the mass as a result of the stage separation in multistage rockets. An example of the orbital injection of rocket is given in Fig. 3b. For the method of injection, represented in this figure, the laying out of powered phase to two parts is characteristic. From the Earth and to point C of trajectory the engines accelerate/disperse rocket and turn/run up it to the transfer ellipse. At point C the engines are turned off. The rocket continues

lift with a decrease in the velocity. At point B (actually in the apogee of transfer ellipse) the engines again are switched on and to rocket is communicated the velocity, which exceeds circular at the given altitude. As a result the satellite is put into orbit, the perigee altitude of which is equal to apogee altitude of transfer ellipse.

Motion of space vehicle beyond limits of the atmosphere.

Coordinate systems. As was already mentioned, the motion of spacecraft just as all celestial bodies within the limits of the solar system, it is determined in essence by the gravitational forces acting on them. The structure of the solar system in many instances makes it possible to present the free flight of spacecraft by that occurring in the gravitational field of one or the other celestial body. This gives the possibility to describe the motion of planets, satellites and other celestial bodies in the systems of coordinates, whose beginnings are placed in the centers of gravitational fields.

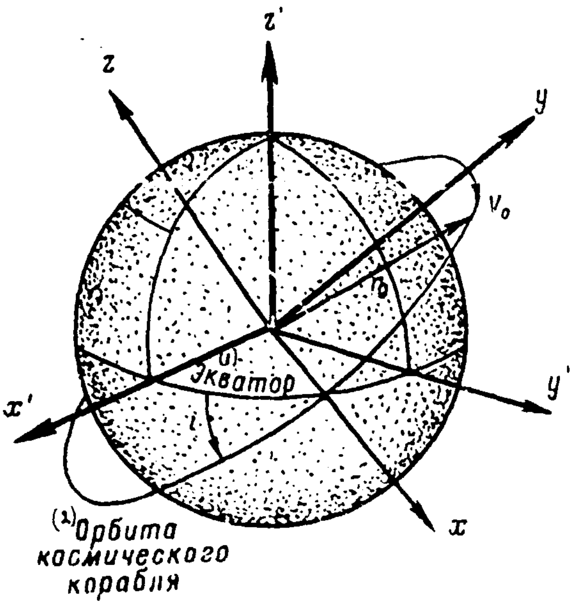

Fig. 4. Geocentric orbital coordinate system.

Key: (1). Equator. (2). Orbit of spacecraft.

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The corresponding systems of coordinates are usually called the name of that celestial body to the center of which they are placed. For example, the heliocentric coordinate system originates in the center of the sun, geocentric - in the center of the Earth.

Fig. 4 depicts coordinate system, whose beginning is in center of Earth. X axis passes through the descending unit of orbit and it lies/rests at the equatorial plane. Z axis is perpendicular toward the orbital plane and is directed along the vector of orbital angular velocity. Y axis is perpendicular to plane x-z. Axes/axles x', y' and z' are cabled to the planet, in this case - to the Earth.

Orbits of artificial Earth satellites. Depending on velocity, heights/altitudes also of the angle of the slope of the trajectory tc the local horizon at the end of the powered phase it is possible to obtain one or the other orbit of artificial Earth satellite. If the final velocity of carrier rocket in the powered phase is equal to circular, then we will obtain the circular orbit (Fig. 5a). If the final velocity of the rocket booster is more than circular for the given injection altitude, then the orbit will be elliptical. In this case the point of the greatest distance of the satellite from the Earth (apogee) will be always found at a higher altitude than the injection point.

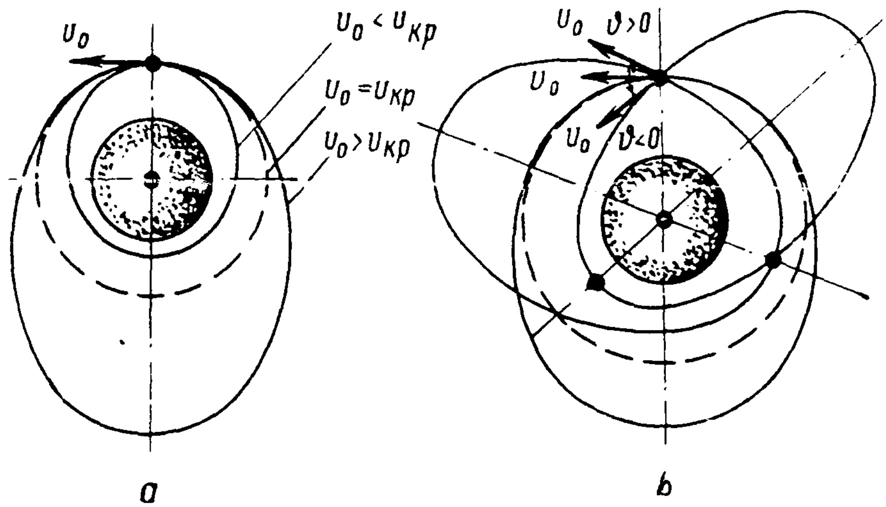

Fig. 5. Change in orbit of the satellite: a) depending on velocity of injection; b) depending on the direction of the vector of speed of injection.

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But if the final speed is less than the circular, then the point of the shortest distance of the satellite from the Earth (perigee) there will always be below injection altitude.

At one and the same speed and injection altitude change in flight path angle to local horizon considerably affects perigee altitude of orbit. Greatest perigee altitude - at the zero flight path angle to the horizon. An increase or a decrease of this angle decreases the perigee altitude and is changed the trajectory of satellite with respect to the Earth (Fig. 5b).

Time of one revolution of satellite around Earth (orbital period) with circular orbit with radius of r

$$T_{\text{кр}} = \frac{2\pi r}{V_{\text{кр}}} = \frac{2\pi}{\sqrt{K}} r^{3/2}. \qquad (4)$$

Time of access of Earth satellite in flight at altitude of 300 km along circular orbit according to formula (4) is approximately 90 min. With an increase in altitude it intensely grows/rises. It is easy to find the altitude, at which the time of the access of satellite is equal to the daily rotation of Earth $T_{\text{кр}}=23$ hour 56 min 4 s. This altitude proves to be equal to 35830 km. The satellite, launched in the equatorial plane eastwards at the given altitude, would be found above one and the same point of the surface of the Earth. This satellite is conventionally designated as stationary, and its orbit - stationary.

If the orbit is not circular, but elliptical, then the orbital period is determined by formula

$$T_{\text{сп}} = \frac{2\pi}{\sqrt{K}} a^{3/2}, \qquad (5)$$

where a - semimajor axis of ellipse; $a = \frac{r_{\text{п}} + r_{\text{а}}}{2}$;

$r_{\text{п}}$ - perigee radius;

$r_{\text{а}}$ - apogee radius.

Let us calculate period of orbit of first Soviet artificial Earth satellite, derived 4 October of 1957 to elliptic orbit. Perigee altitude of it is equal to 228 km, apogee - 947 km. This means that

$$a=\frac{2R_{з}+H_{п}+H_{а}}{2}=6959\ км;$$

$$T_{сп}=\frac{2\pi}{631}\cdot 6959^{3/2}=5770\ \overset{(1)}{сек}=96{,}2\ \overset{(2)}{мин}.$$

Key: (1). s. (2). min.

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This result coincides with the period of orbit of first Soviet artificial Earth satellite in the beginning of its motion. During calculations according to formulas (4) and (5) radii are taken in the kilometers, and the access time is obtained in the seconds.

Artificial Earth satellites at present play an important role in the solution of many scientific problems. Satellites are utilized for the communication, the transmission of television programs, navigation, weather reconnaissance, the collection of scientific data concerning the Earth, etc. Furthermore, in the USA satellites are used also for the solution of military problems: there have launched into space a number of "spy satellites", which, as it is noted in the foreign press, produce photographing of areas of the surface of the Earth important in military sense.

Manned satellites, which will fulfill a number of completely new

functions, such, for example, as rescue of crews in space, maintenance/servicing and repair of different space vehicles, will find an even wider application.

Control of satellite in track- out phase in orbit actually is reduced to control of carrier rocket. After the injection of satellite into calculated orbit to it are presented the following requirements: stable motion in orbit and minimum deviation from it. It should be pointed out that the orbit of satellite will be constant, if it is located at such an altitude where aerodynamic drag is very low, and, furthermore, to satellite is imparted a sufficient speed.

Effective lives of satellites depend on orbit altitude, on which they are launched.

Navigation of satellites in orbit requires determination of a series of necessary data for calculating their position. After the appropriate observations, which make it possible to calculate actual course and speed of flight vehicle, is produced the calculation of the necessary changes in the course and the speed for the completion of the maneuvers, provided for by assignments for the flight. Navigational observation is most effectively accomplished/realized by ground tracking stations.

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However, with the aid of the onboard instruments of person, who is

located on the satellite, it can determine the parameters of orbit of satellite, also, at the loss of connection/communication with the Earth. If the instrument, which makes it possible continuously to control the ground-based position of the geographical point of satellite during its determination in orbit, is located on board the satellite, then because of it it is possible to obtain all necessary data, which require for calculating the orbital parameters.

Flights of spacecraft. If to the spacecraft there will be imparted such a speed that its kinetic energy

$\frac{mV^2}{2}$ will prove to be the equal potential energy $\frac{mK}{r^2}$,

then the ship will leave the gravitational field of the Earth. This speed, as noted, is called parabolic velocity, speed of "release" or planet escape velocity. It is equal to 11.1859 km/s (it is usually accepted as 11.2 km/s).

Fig. 6 shows possible regions of flights of spacecraft in coordinates "altitude-speed". The region of flights of ballistic missiles is shown here.

Region of flights of spacecraft is from below limited by curve of so-called aerodynamic "barrier". On top it is also limited to the curve, which is called temporary/time "barrier". By the time "barrier" in this case there is the duration of flight (taking into

account return to the earth) not more than 10 years. To the left this region is limited curved orbital velocity, and to the right curved, that limits the region, where the accelerations do not exceed 2 g. Furthermore, the region in question is divided in the sections, in each of which it is most profitable to utilize the specific type of rocket engine. This separation is carried out in accordance with that speed range, which is optimum for the engine of this type.

Let us examine possible orbits of interplanetary flights and will analyze some considerations on their selection.

Let us recall that orbital planes of planets either coincide with plane of orbit of Earth or insignificantly they diverge from it. Planets orbit, close to the circular.

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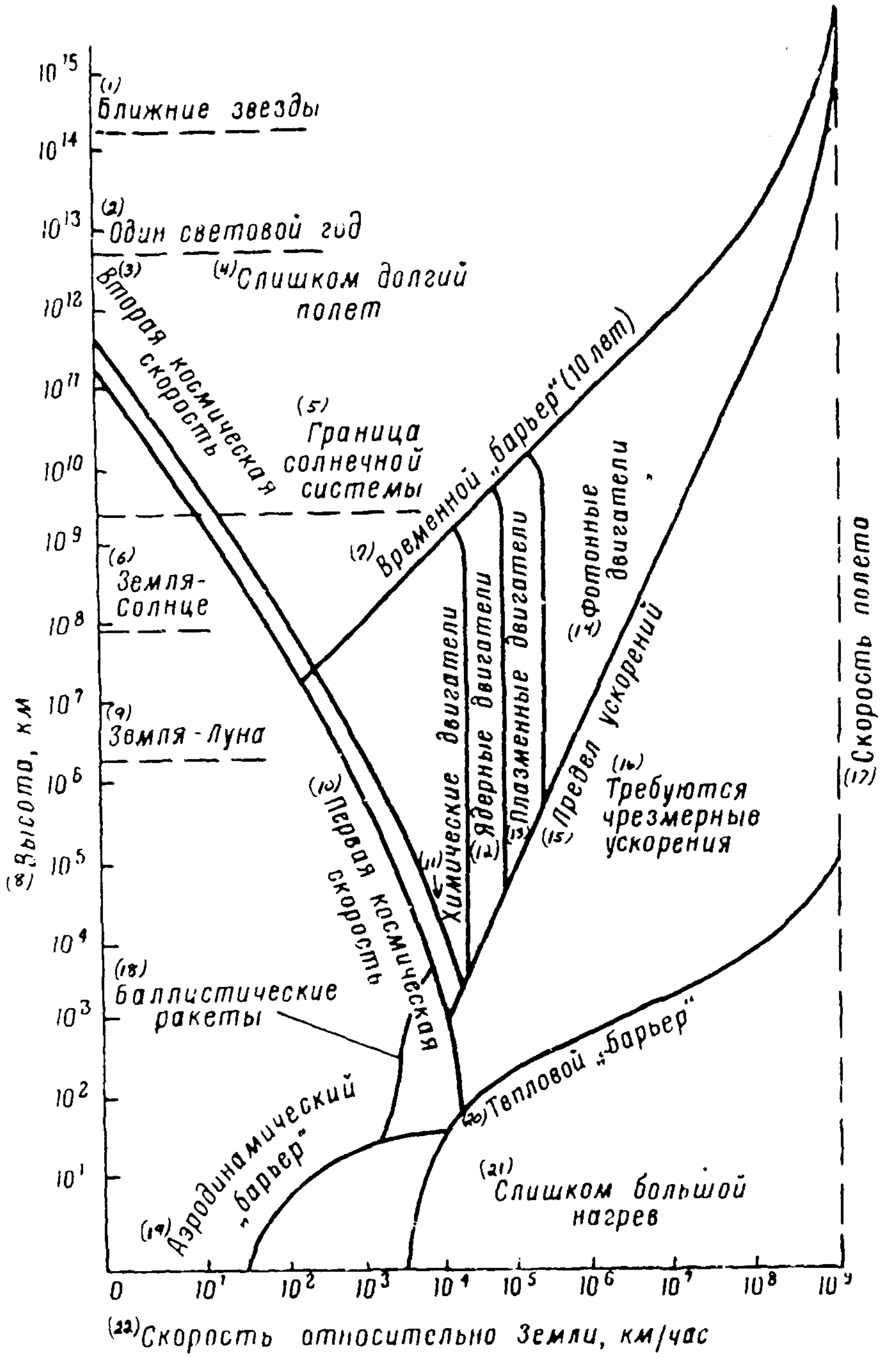

Fig. 6. Regions of flights of spacecraft.

Key: (1). Near stars. (2). One light year. (3). Planet escape velocity. (4). Too long a flight. (5). Boundary of solar system.

(6). Earth-Sun. (7). Time "barrier" (10 years). (8). Altitude, km. (9). Earth-Moon. (10). Orbital velocity. (11). Chemical engines. (12). Nuclear engines. (13). Plasma engines. (14). Photon engines. (15). Limit of accelerations. (16). Excessive accelerations are required. (17). Flight speed. (18). Ballistic missiles. (19). Aerodynamic "barrier". (20). Thermal "barrier". (21). Too great a heating. (22). Speed relative to Earth, km/h.

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On them the gravitational acceleration, caused by the gravitational field of the sun acts (Fig. 7). There are regions of the space, where the predominant effect exerts the gravitational field of planet. This is - the sphere of influence of planet. By sphere of influence of planet is implied the part of the space, in which during the analysis of the motion of small body with the planet escape velocity by the fundamental center of gravity should be counted the planet, but not the sun.

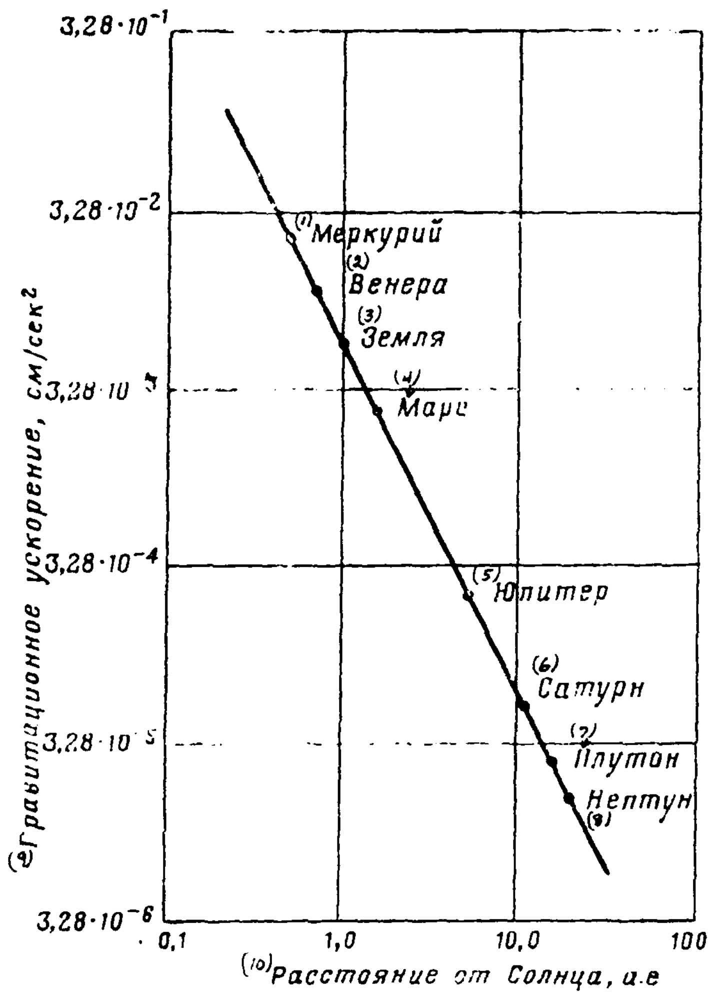

Fig. 7. Gravitational pole of the Sun (along horizontal axis plotted distance from sun in astronomical units).

Key: (1). Mercury. (2). Venus. (3). Earth. (4). To fog. (5). Jupiter. (6). Saturn. (7). Pluto. (8). Neptune. (9). Gravitational acceleration, cm/s². (10). Distance from sun, AU.

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In the first space interplanetary flights of man, undoubtedly, there will be used minimum energy orbits, on which the attracting

force of the sun will play main role during motion of spacecraft.

Flight of interplanetary spacecraft will consist of a number of stages, each of which will require independent approach to solution of navigational problem taking into account effect of gravitational poles of the Sun, and also departure planets and designation/purpose.

First stage - start of ship. In this stage the ship starts from the surface of the Earth (or departure planet), it is accelerated/dispersed and departs from the Earth's gravitational field, which exerts in this section the predominant influence aboard the ship. At the termination of the work of engine installation to ship is communicated the speed, sufficient for the ejection beyond the limits of the sphere of the attraction of the Earth. The predominant effect aboard the ship in this stage exerts the gravitational field of the Earth.

Second stage - flight in the middle section. In this stage the ship passes the greater part of the distance between the planets along the free trajectory or with the low thrust. The task of navigation in this case is the determination of the position of ship in the trajectory, along which it in the preset time will leave into the area of destination planet. On the second stage it is necessary to calculate initial data for the correction maneuvers and to fulfill these maneuvers. The predominant effect aboard the ship in this stage exerts gravitational pole of the Sun.

Third stage - flight in section of drive. In this stage the ship passes in orbit, along which it emerges to the destination planet. The predominant effect on the flight here exerts the gravitational field of this planet. The ship enters into the atmosphere of the planet in limits of the landing corridor, the form and dimensions of which are determined by the parameters caused by characteristics of the planet and spacecraft.

Fourth stage - the return to the earth (destination planet).

What are such distances of stages of flight of spacecraft during interplanetary flight? The first stage is determined by the radius of the sphere of the attraction of the Earth, which is approximately 925,000 km.

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The length of the second stage (heliocentric transfer orbit to the nearest of the planets) many times exceeds this distance and composes hundreds of millions of kilometers. Thus, it is possible to consider that basic part of the middle trajectory phase is located out of the limits of the sphere of influence of the Earth.

If there are known origin coordinates and flight speed of spacecraft, and also characteristic of gravitational field, in which ship moves, then to theoretically accomplish space navigation is

possible without the use of external reference points, cosmic bodies or special beacons. If the system acts without the use of external reference points, the inertial system of navigation for determinirg the orientation of ship must be used on board the ship.

Position and flight speed of spacecraft can be determined by direct measurement by their astronomical methods, being based on knowledge of laws of planetary motion relative to sun. The stars or the planets, utilized as the reference points in similar measurements, are actually the navigation aids of the same character as reference points and the beacons, utilized in the sea or air navigation. The parameters of the trajectory of spacecraft can be measured also with the aid of the ground tracking stations. It is possible to assume that in the future such stations will be established/installed on one or several celestial bodies.

Flights of space vehicles to the Moon became already a reality. The Moon is the closest to the Earth celestial body of the solar system. It is natural Earth satellite. Average distance from the Earth to the Moon is equal to 384,400 km. In the perigee the distance to the Moon is equal to 363,000 km, in the apogee - 405,500 km. The mass of the Moon composes 0.01 of mass of the Earth, diameter - 0.27 of diameter of the Earth, space - 0.02 of space of the Earth, density - 0.61 of Earth density. The surface gravity of the Moon is equal to 1.62 m/s^2, i.e., it is approximately six times less than the terrestrial. Consequently, the weight of any object on the Moon is

six times less than that on the Earth. The Moon always is turned to the Earth by one and the same side. This is explained by the fact that it rotates around its axis with the same period with which it turns around the Earth.

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For the flight to the Moon is required the considerably smaller energy consumption of carrier rockets and it is less time than for the flight to the more distant planets. A total of several days are necessary in order to complete flight to the Moon, while for any other interplanetary flight/passage they will be required months and years.

Earth and Moon frequently considers as single dynamic system. The general/common center of mass of this system, or the so-called barycenter, is arranged/located on the line, which connects centers of both bodies, and is located from the center of the Earth approximately on 5000 km.

In contrast to Earth Moon is almost deprived of atmosphere. Consequently, during the landing on the Moon there it is not possible to extinguish the energy of ship by its transformation into heat, as is done with descent in the atmosphere. Therefore, aboard the ship it is necessary to have a special device for the braking and shock absorption at the moment of landing.

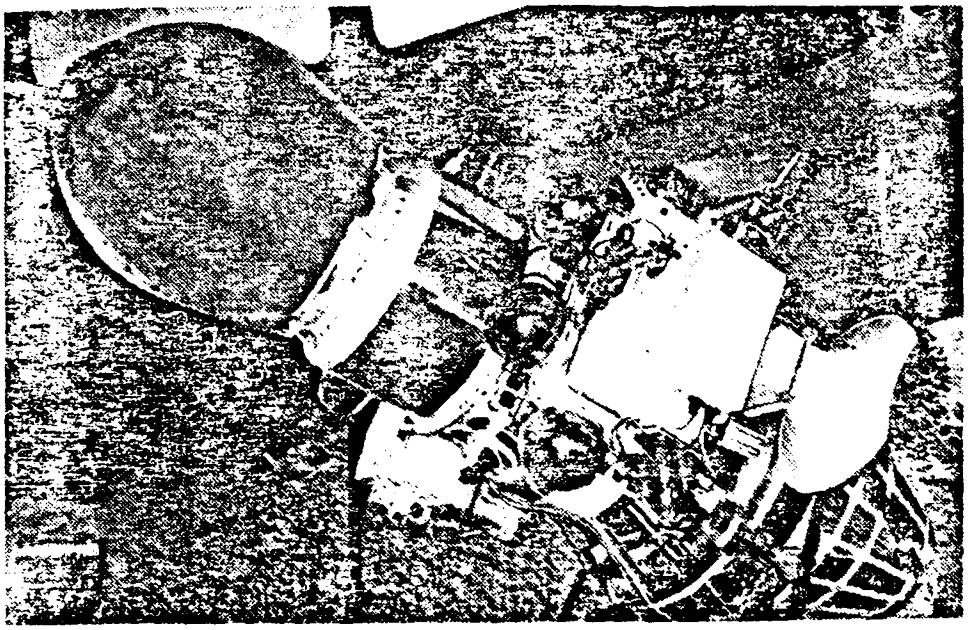

Fig. 8. Soviet automatic station "Luna-9" with instrument compartment and retro-engine installation, for the first time in the world completed a soft landing on the Moon.

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The first in the history of humanity space vehicle which completed a soft landing on the Moon, was, as noted above, the automatic station "Luna-9" (Fig. 8). On 3 February, 1966, it reached the Moon and was lowered to its surface in the area of Oceanus Procellarum. With the aid of the television equipment the station transmitted to the earth the image of lunar landscape.

On 24 December, 1966, to Moon (also into area of Oceanus Procellarum) completed soft landing Soviet automatic station "Luna-13". Its scientific instruments conducted valuable measurements

regarding the properties of lunar soil, intensity of corpuscular radiation, etc. The transfer of the television image of lunar panorama was realized also.

During June 1966 to Moon completed landing American space vehicle "Surveyor-I" (Fig. 9). Its weight with the launch was 995 kgf, including the weight of the braking RDTT [Solid-propellant rocket engine] - 624.6 kgf. After landing on the Moon the apparatus weighed 283 kgf.

The apparatus "Surveyor-I" consisted of an aluminum framework/body, made from hollow tubes. To the framework/body were fastened two omnidirectional antennas, two containers with the electronic equipment and the equipment of the electric power supply system, the braking RDTT, three vernier ZhRD [Liquid propellant rocket engine], landing equipment and other equipment. Braking RDTT created thrust of 3.6-4.5 T. Three vernier ZhRD operated on hypergolic fuel; the engine thrust could be regulated in the range 13.6-47 kgf. Landing chassis/landing gear of apparatus had three struts with the supports of honeycomb construction/design. Struts were equipped with aircraft type hydraulic shock absorbers, and also with telescopic thrusts, which provided the unfolding of the landing gear after the launch of a vehicle to the flight trajectory to the Moon. Apparatus was designed for the landing with a vertical component of speed up to 6 m/s. The electric power supply of onboard equipment was produced by solar cells and silver-zinc batteries. Solar cells provided the

maximum power of 89 W. They recharged silver-zinc batteries. There was aboard also a system of the flight control and the orientation, into which entered the sun sensors for the automatic search for the sun and radars of different designations/purposes.

After lunar impact the instruments installed on the apparatus took scientific measurements, which were transmitted with the aid of radio equipment to the earth.

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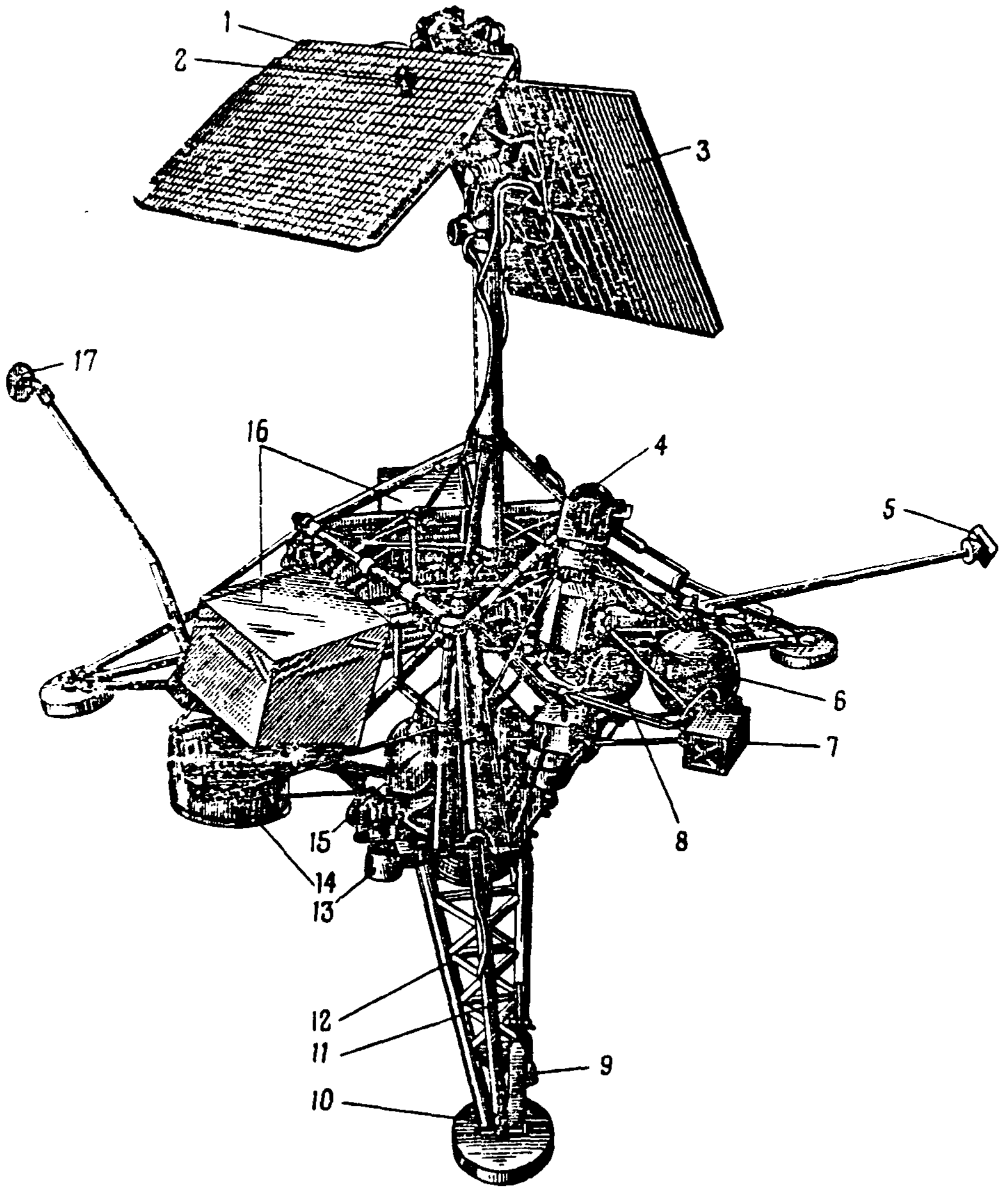

Fig. 9. American space vehicle "Surveyor-I": 1 - solar blade; 2 - sun sensor; 3 - high gain antenna; 4 - television camera; 5, 17 - omnidirectional antennas; 6 - tank with compressed helium; 7 - auxiliary battery; 8 - tank with compressed nitrogen; 9 - jet nozzles of orientation; 10 - heel; 11 - damper; 12 - landing gear strut; 13 - vernier ZhRD; 14 - antenna of radio altimeter; 15 - fuel tank for

vernier ZhRD; 16 - containers with radio-electronic equipment.

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Being transmitted to the earth were also television images of the lunar surface.

Rating/estimating flights of Soviet stations "Luna-9" and "Luna-13" and American apparatus "Surveyor-I", it is possible to conclude that they demonstrated effectiveness of systems, which ensure soft landing on surface of Moon, and also effectiveness of onboard systems, including onboard engines. The obtained information about the bearing capacity of lunar soil, the temperature conditions to the surface of the Moon and some others data expanded knowledge about this planet.

During September 1966 was launched to Moon American space vehicle "Surveyor- II". The weight of the apparatus was 999.7 kgf. According to the calculated flight program approximately 63 hours after launching the apparatus had to complete soft landing on the Moon. For decreasing the vertical velocity to 9.8 m/s had to be switched on three vernier ZhRD. However, one of the engines was not switched on, and the apparatus began disorderly to rotate. All attempts to stabilize it with the aid of the controls of nozzles of the orientation system, and to also switch on the Vernier rocket were finished by failure. Later for the stabilization there was switched on the braking RDTT. The communication with the apparatus was

interrupted for 30 s after this.

During January of 1968 on the surface of Moon there completed soft landing of the American space vehicle "Surveyor-VII". The television images of lunar surface were transmitted with the aid of the television equipment installed on it, to the earth.

Foreign press reports that the USA intends in 1970 to land on the Moon cosmonauts, for which there are intensely conducted tests of the "Apollo" spacecraft.

Docking of space vehicles. In the contemporary space flights the task of rendezvous in space of two space vehicles and realization of their mating extremely importantly acquires importance. It is caused, on one hand, by considerations of the guarantee of minimum power expenditures during the launching of the apparatuses, and, on the other hand, by the need for the creation of orbiting scientific space stations, delivery to them of loads, replacement of crews, repair in space, etc.

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By term "docking" there is implied the realization of the mechanical contact between two space vehicles. Docking is the final rendezvous phase in orbit. It consists of the rendezvous maneuver, creation of contact and final rigid connection.

Development of the docking system, as it is reported in the American press, was begun from the determination of general principles of its accomplishing. First of all it was necessary to establish/install, in what section of space vehicle it is most expedient to make an attachment/docking point. It was decided to utilize a forward hold for the docking, since in this case it is possible to accomplish visual control of the docking procedure and directly to control it.

Another task consisted in selection of type of target - active or passive. Active target is capable of approaching an object of docking as a result of the correction of its own orbit, use of a radio beacon or flashing lights and bears on itself the important parts of the docking system. Passive target cannot approach an object of docking and does not bear on itself attachment/docking points.

For conducting experiments in the USA there were selected the manned satellite "Gemini", which served as pursuit vehicle, and rocket booster stage "Agena", which served as purpose[1].

FOOTNOTE[1]. In the Soviet literature it is accepted the name the interceptor - the active space vehicle, and target - passive. ENDFOOTNOTE.

Despite the fact that the use of a passive target during the adjustment of the operation of docking expands the possibilities of pursuit vehicle, it was it is nevertheless decided to place basic part of the docking system on the missile-target. This is caused by the fact that payload weight, which can put into orbit this carrier rocket, as "Titan-II", is limited. In the furture, it is proposed to convert the spacecraft into a completely active system, which has the capability of docking with the passive target.

Was investigated also possibility of repeated accomplishing of docking. The system, intended for the one- shot docking, is sufficiently simple. But specialists went for its notorious complication connected with repeated application.

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For an evaluation of power expenditures during docking and selection of the damping system, the docking procedure was investigated by simulation. In the final analysis the docking system, which consists of the section of the rendezvous of space pursuit vehicle and reciprocal tail butt cone, established/installed on the missile-target, was developed.

Docking is accomplished in three stages: first is fulfilled orbital rendezvous of two apparatuses, then rendezvous maneuver, after which performs strictly docking.

Orbital rendezvous. Let us examine how it occurs. For simplification one of the apparatuses, which is injected into orbit, let us accept for the supporting/reference, another - for the interceptor. For the realization of rendezvous must be sustained this sequence of the maneuvers: change in the orbital plane, transition/transfer from one orbit to another, that is located in the same plane, acceleration or deboost along the orbit and final.

The easiest method, the using of which it is possible to carry out rendezvous of two apparatuses, is the immediate putting of the interceptor into orbit of the apparatus located in flight. Control of the carrier rocket, which injects the interceptor for a rendezvous, is accomplished by the appropriate equipment from the place of start or complex of the tracking stations, placed along the course. In order to compensate for the errors, which can arise in the process of the work of control system of carrier rocket, usually are provided for the maneuvers of apparatus in the final trajectory, before the rendezvous.

Flight trajectory of the carrier rocket, which places in orbit interceptor, consists of several sections: removal, correction, search and target designation and finally homing. This last section of active guidance leads to the rendezvous of satellites. Supporting

apparatus is found in known orbit, and tracking continuously is accomplished/realized after it, i.e., the parameters of its orbit are determined. These parameters are introduced into the memory unit of the guidance system of the interceptor. The starting time of the interceptor is planned with the high precision/accuracy with respect to the moment/torque of the passage of the supporting apparatus above it.

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Is started interceptor in one plane with the trajectory of supporting apparatus, which provides the rendezvous of supporting apparatus after the accomplishment of the correction maneuver. Fig. 10 shows the schematic of the rendezvous of two apparatuses by the method indicated.

More common is the case of the rendezvous of apparatuses in orbit, when interceptor is placed in orbit, which is located not in that plane, in which there is located an orbit of supporting apparatus. This requires maneuvers for changing the plane of orbit and maneuvers in the orbital plane. Let us assume that the support apparatus is found in this orbit, and the interceptor must be launched from the same place of start, as supporting apparatus. As a result of rotating the Earth the orbit of supporting apparatus will be designed on the surface of the Earth as extended curve (Fig. 11). The place of start will be located in the plane of the orbit of supporting apparatus one time in 12 hours. Exception is only the case, when apparatus is launched in orbit with the zero angle of inclination/slope from the place of start, situated on the equator.

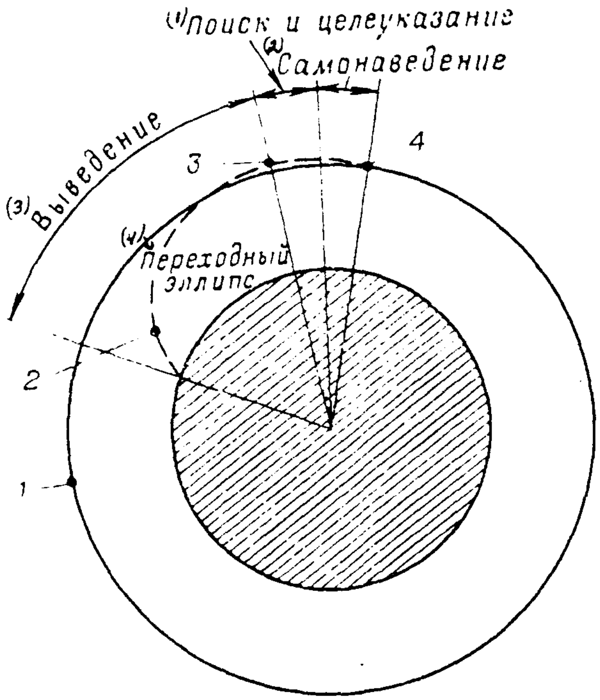

Fig. 10. Diagram of the rendezvous of apparatuses in orbit: 1 - position of supporting apparatus at the moment of launching of the interceptor; 2 - engine cutoff of carrier rocket; 3 - correction; 4 - orbital rendezvous.

Key: (1). Search and target designation. (2). Homing. (3). Injection. (4). Transfer ellipse.

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In this case the supporting apparatus will have a latitude of the point of the arrangement of the place of start twice during each revolution of apparatus. If the angle of orbit inclination is equal to the latitude of the point of the arrangement of the place of start, ground track on the surface of the Earth will oscillate in the vicinities of the latitude of the place of start.

Rendezvous maneuver differs from other maneuvers of space vehicles in terms of low relative speeds and in terms of distances. Target range at the moment of the beginning of approach is usually 30-500 m, the initial velocity of approach - 1.5-3.0 m/s. The initial conditions of approach depend on the precision/accuracy of the work of the guidance and navigation systems of ship in the finite segment. For example, the rate of closure of American spacecraft "Gemini" with the rocket "Agena" was accepted as 0.45 m/s, and the lateral displacement - 0.3 m.

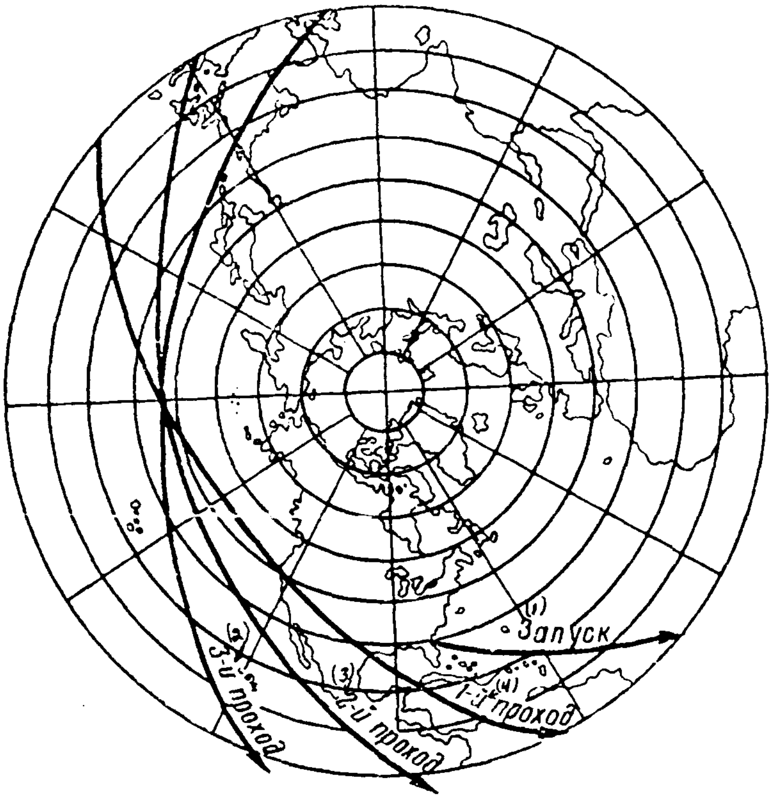

Fig. 11. Polar diagram of orbital tracks (T=90 min).

Key: (1). Launching. (2). 3rd passage. (3). 2nd passage. (4). 1st passage.

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It is understandable that if the rate of closure up to the moment/torque of contact is excessively great, then there appears the danger of the damage of the butting apparatuses as a result of impact with contact. As practice shows, the engine installation necessary for the rendezvous maneuver must provide low accelerations, and therefore it can have an insignificant fuel reserve.

It is necessary to indicate that the butting apparatuses are actually found in different orbits. Their relative motion affect the different values of the gravitational acceleration, which, however, are such small that they usually disregard them. With the realization of docking required directions of thrust can be most varied, and in the general case should be had the capability to create positive and negative thrust along all three body axes. For this in the interceptor it is necessary to establish/install a large quantity of engines or to have an orientation system, capable of rapidly turning/running up it in the prescribed/assigned direction.

As it is indicated in the foreign press, control forces and moments during docking can be created not only by rocket engines, but also by electromagnets. An analysis shows that the installation on the butting objects of the electromagnets, which possess the appropriate characteristics, makes it possible to obtain the magnetic fields of a sufficient intensity, capable of attracting/tightening apparatuses of each other and of aligning their axes.

An example of the engine installation, intended for maneuver accomplishment of approach, is orientation system and maneuvering of spacecraft "Gemini". System consists of sixteen jet nozzles of engine installation, arranged/located in the transition piece, connected with the section of entry into the atmosphere (Fig. 12). Eight nozzles are intended for the maneuvering and eight - for the orientation. Of eight nozzles of orientation four serve for pitch control and four -

on the yaw angle.

Practice of ground-based simulation and flight experiments attest to the fact that cosmonaut is capable to successfully fulfill docking only during visual orientation. In this case the rate of closure is determined with the maximum precision/accuracy, if the angle of the image of target composes 80-90°.

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At a distance of 3 m to the target the cosmonaut can note approach and determine speed, beginning from value on the order of 5 cm/s. As it is indicated in the foreign press, the practice of simulation showed that after a certain number of trainings the cosmonauts produce the necessary skills in the determination of rate of closure, which makes it possible for them to satisfy the prescribed final conditions for maneuver.

If an automatic guidance system is utilized on a space vehicle, then for the realization of docking the onboard computer produces control signals which enter the system of control and orientation.

The determination of possibilities of the docking mechanism is the main thing in the selection of the guidance method. If it allows a change in the rates of closure over a wide range and in this case it is not necessary to rapidly accomplish/realize an approach, i.e., sense to previously establish/install initial relative rate of closure

and to further accomplish/realize docking. If approach time not at all limited, an the final speed must be maintained/withstood sufficiently accurately, then it is expedient at the very beginning to establish/install the required final speed and to support with its constant.

Docking is accomplished in three stages.

First stage - off-normal contact.

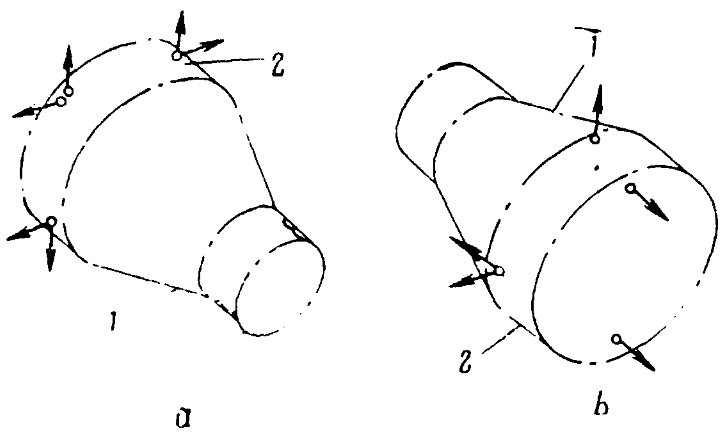

Fig. 12. Engine installation of system of spacecraft "Gemini": a) for changing orientation; b) for maneuvering; 1 - section of entry into the atmosphere; 2 - transition piece.

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It occurs with a pulse, i.e., at the moment of contact the linear and angular velocities of objects in the constant/invariable relative attitudes change instantly as with the impact/shock.

Second stage - joint motion of butting objects with reduced number of degrees of freedom. The mutual displacement of objects with the maintaining of contact occurs in this stage.

Third stage - the final rigid connection. The locks operate, and the docking objects are converted into one rigid system.

It should be pointed out that docking was already virtually carried out. In 1966 there was implemented an experiment on the

docking of American spacecraft "Gemini" with the rocket "Agena". For this ship it was conducted/supplied to the rocket approximately on 0.5 m so that both apparatuses continued to orbit, retaining the established position during 2 min, after which from the Earth there was reported the readiness for the reception of telemetry data. Then cosmonauts at the relative forward velocity of 0.23 m/s performed docking. Immediately after it, utilizing an orientation system of rocket "Agena", they produced the turn of the butted apparatuses on 90° in the horizontal plane. Turn occurred normally, and the connection proved to be sufficiently rigid. However, the rotation of apparatuses was begun after a certain time. When rotation began to threaten flight safety, cosmonauts were were forced to be undocked with the rocket. According to the communication/report to the foreign press, short circuit in the electrical circuit of the orientation system and maneuvering of ship was the reason for the rotation of apparatuses.

This experiment confirmed the effectiveness of the used rendezvous maneuver of space vehicles, and also the relative simple mechanism of docking, which was selected from a number of others only after as a result of ground-based simulation it was possible to determine the possibility of guaranteeing low relative rates of closure. Docking maneuver was satisfied by the craft commander by hand on the base of visual information. The engine installation of ship made it possible to change directions of thrust along all three axes. This made possible to cosmonauts to avoid the considerable

turns of ship and not to lose out of sight rocket.

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At the end of October of 1967 in the Soviet Union for the first time there was produced automatic docking in orbit of two artificial Earth satellites "Kosmos-188 and "Kosmos-186[1].

FOOTNOTE[1]. Satellite "Kosmos-186" was launched into orbit on 27 October, "Kosmos-188" - on 30 October, 1967. ENDFOOTNOTE.

Satellites were equipped with special rendezvous systems with the units of docking (on one satellite rod, on other - receiving cone). For the realization of docking the satellites produced a series of complicated maneuvers in outer space. Were automatically carried out mutual search, approach, contact, and then - rigid docking of satellites. The distance between the satellites at the moment of the injection of satellite "Kosmos-188" into orbit was approximately 24 km, and the relative speed of their motion was on the order of 25 m/s. The processes of search, approach and docking were accomplished/realized with the aid of the special radio equipment and the computers, established/installed on board the satellites. Television equipment carried out a transfer to the earth of the image of the butted apparatuses and process of their conjointing. In the joined state the satellites were 3 hour 30 min. After a certain time after the joining both satellites with the aid of the onboard engine installations were transferred to different orbits. During

April of 1968 the automatic docking of satellites was repeatedly produced in orbit. In this case the range of the beginning of the approach was 5 km, and the relative rate of closure of the satellites - 30 m/s. When the distance decreased to several hundred meters, the engine plant of low thrust operated, and the relative speed decreased to 0.1-0.2 m/s.

Hardly there is sense to speak of the fact that the implementation of automatic docking is a task considerably more complicated than the production of visual docking. This is the truly salient scientific experiment.

Docking mechanisms have a special importance for docking in space.

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They must decrease the difference in the speeds of spacecraft to zero; to scatter kinetic energy; to provide the mechanical connection/communication between the ships after contact; to create the repellent forces, which make it possible for ships, if this it is necessary, immediately to be disconnected; to provide the transfer of electrical signals, pumping fuel/propellant, transfer of loads and finally the transition/transfer of cosmonauts from ship to ship.

In foreign press it is noted that in the USA there are developed docking mechanisms of different designs.

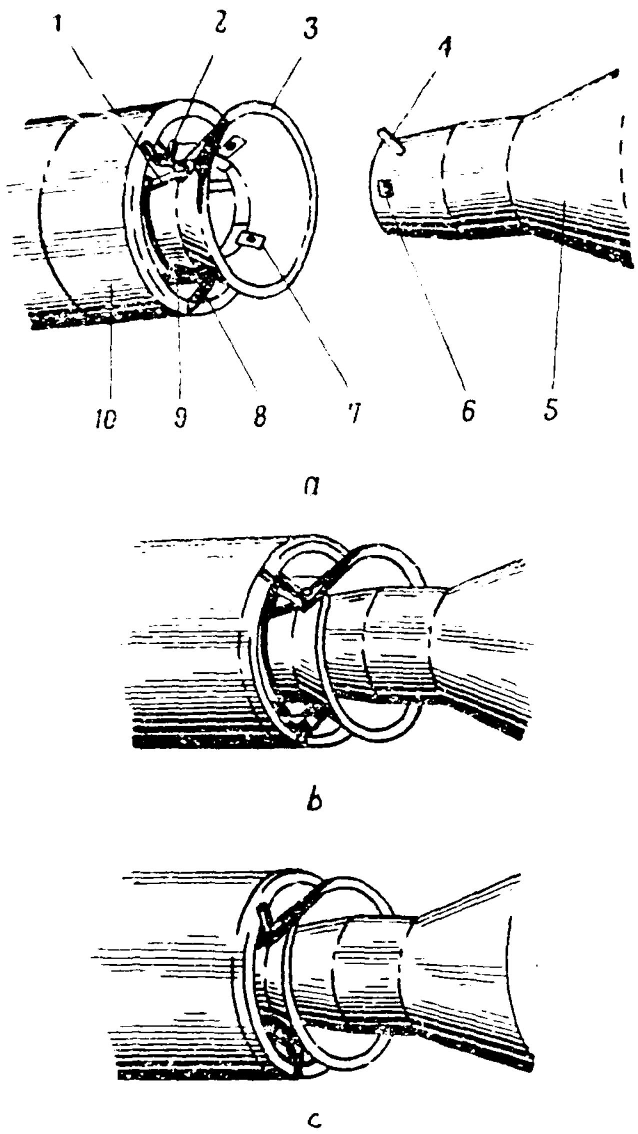

Fig. 13. Docking procedure of spacecraft "Gemini" and rocket "Agena": a) approach; b) docking; c) connection; 1 - upper angular longitudinal damper; 2 - upper lateral damper; 3 - butt cone; 4 - attaching rod; 5 - spacecraft "Gemini"; 6 - slot for latch (one of three); 7 - latch (one of three); 8 - lower lateral damper; 9 - lower longitudinal damper; 10 - rocket "Agena".

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We will point out only one of them, which was used for the docking of spacecraft "Gemini" with the rocket "Agena" (Fig. 13). This mechanism consists of the receiving tail conical attachment/docking point, establishedfinstalled on the missile-target, and the fixed/recorded unit in the forward hold of spacecraft. During the docking the forward hold of ship is introduced into the cone of the receiving unit, which is fed back and with the aid of a system of shock absorption (dampers) deadens the shocks. Then three lock springs fall into the appropriate seats in the forward hold. Thus the preliminary connection is attained. In this case into the slot in the upper part of the butt cone enters the fixing rod, fastened/strengthened in the forward hold, which prevents the possibility of the mutual rotation of the butting objects. Only the rigid connection of the spacecraft and rocket occurs after this. The forward hold, which continues to be advanced under the effect of the spring of latches, by the leading edge is abutted against three catches, fastened on the base of the butt cone. The butt cone itself is moved to the end position, being abutted against powerful supports on the collar of transfer missile bay. In this final state, fixed by latches, both butting objects must maneuver as one whole.

Loads during maneuvering are transmitted directly to elements of construction/design of objects and butt cone, passing/avoiding dampers of damping system. The damping system consists of three lateral and four longitudinal dampers, evenly arranged/located by three groups

around the base/root of butt cone.

Together with the docking procedure indicated there is conducted work on the creation of universal docking mechanisms for use aboard all ships which can participate in the rendezvous. Creation in the process of these work of the mechanism of docking at a high relative speeds is one of the tasks. This will make it possible to decrease requirements for the system of control and maneuvering of spacecraft and to reduce the fuel consumption per orbit at the moment of docking.

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As correctly it is indicated in foreign press, during docking at high relative speeds orbit of target will undergo some disturbances, which, possibly, it is necessary then to compensate. Furthermore, if one must carry out docking with the target, whose weight is compared with the weight of the interceptor, during capture the oscillations/vibrations of the butting objects can arise. The oscillations/vibrations of target can be eliminated by different methods, in particular, by the alignment/levelling the position of ships or by the application/appendix of the engine thrust.

ENTRY INTO THE ATMOSPHERE.

Safe landing of the spacecraft on the surface of a planet is the completion of space flight along an interplanetary trajectory or along the orbit of an artificial satellite. For planets which have an

atmosphere, this task is reduced to the permission/resolution of the problems of g-forces, aerodynamic heating, control of the time of reaching/achievement of planet and research of landing place. In the absence of atmosphere remain only the problems of g-forces, control of the time of reaching/achievement of planet and search for landing place.

The ship which approaches atmosphere of planet from outer space or which descends from orbit of artificial satellite possesses a arge reserve of energy. This energy consists of the kinetic energy caused by the speed of ship, and the potential energy caused by its position relative to the surface of the planet. As before any body entering the dense layers of the atmosphere with hypersonic speeds, a powerful wave appears before the spacecraft in its nose section. The density of the gas and its temperature in the shock wave sharply grow/rise. In proportion to the "immersion" into the denser layers of the atmosphere, the ship is increasingly more heated and its speed as a result of aerodynamic braking continuously decreases. In this case the kinetic energy of ship is converted into the heat. If the entire heat, which is formed in this case, departed for heating of ship, then it it would prove to be sufficiently for the total evaporation of the design of the ship. However, in actuality, as is evident from the cases at least of the incidence/drop in the meteorites to the earth, the significant part of the energy is abstracted/removed into the surrounding space. Heat is taken away by shock waves.

Heat transfer by shock waves is the result of interaction of the molecules of the gas, which surrounds the flying body. A compressed to the high pressure and heated to the high temperature layer of gas, in which occurs the process of interaction of particles, is limited from the front by shock wave front. Shock wave will move away far in the atmosphere, in different directions from the ship, and is left the wide trace, formed by the heated gas. In it is included basic part of the heat, which is isolated upon the entry of ship in the atmosphere. The heat flux, which reaches the surface of ship, enters from a layer of the compressed air mainly due to the friction.

Part of the Heat, which exits in the atmosphere, is directly proportional to intensity of shock wave: the stronger wave, the less quantity of heat is transmitted to ship as a result of friction. The strongest shock waves appear when the leading (nose) section of the body is blunted. Therefore, the usually blunted streamlined shapes, but not elongated, which are classical in aerodynamics of low subsonic and supersonic speeds are given to the ships, intended for the descent in the atmosphere, (Fig. 14).

For the removal of heat which is formed as a result of aerodynamic heating, from construction/design of ship several methods at present are utilized.

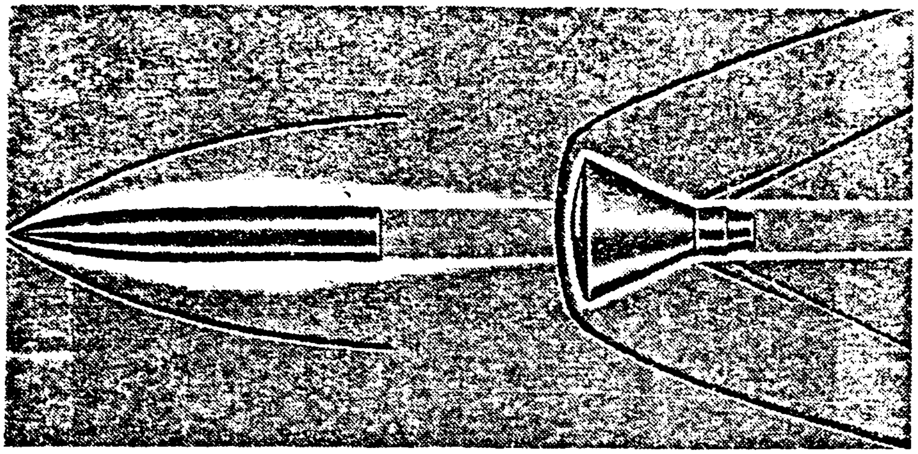

Fig. 14. Pattern of aerodynamic heating of bodies of different form, depending on the intensity of the shock wave (dark curves).

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If there is accomplished ballistic descent in the atmosphere or entry into the atmosphere at large flight path angles, the ship for a short time interval reaches lower, denser layers of the atmosphere. Braking of the ship is accomplished in this period. Consequently, the heat flux, or the quantity of heat which enters the ship per unit time, will be very large. In this case it is expedient to utilize a ship with the strongly blunted nose section and the sufficiently thick heat shield, capable of actively absorbing heat. The thickness of the protective layer, which absorbs heat, is selected by such that the temperature of skin/sheathing would be limited by the value, permitted for the selected material.

Upon the flat entry into the atmosphere with the use of lift there will be required more time. Here braking apparatus in essence is accomplished at the very high altitudes. Since the atmospheric density is low at such heights/altitudes, heat flux will be also small. It can happen, that it in the final analysis will be equalized with the heat flux, emitted by the surface of ship. In this case it is possible to utilize a method of dissipation of heat with the aid of the radiative cooling of the surface of the ship, covered with thin metal covering.

However, in the opinion of some foreign specialists, simplest resolution of problem of aerodynamic heating upon entry into the atmosphere is now the use of protective coating from heat-insulating

layers of fiberglass and other similar to it materials. As a result of intense heating the outer layer of the heat shield melts and evaporates. The vaporized material slows down heat transfer from the shock wave to the ship.

Besides problem of aerodynamic heating, there is also problem of g-forces, whose solution in certain cases can prove to be more difficult task. For decreasing the g-forces during the braking to the values, which can be transferred by man, is utilized lift. This leads to a decrease in the vertical rate of descent and to the elongation/aspect ratio of the path of flight vehicle to the planet, and, consequently, to a decrease in g-force.

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Lift is utilized sometimes even with high permissible overloads for decreasing the heat fluxes, and also for control of the time of reaching/achievement of planet and of the position of landing place. Its lift-drag ratio is the fundamental aircraft characteristics, which possesses lift. Spacecraft with the value of the lift-drag ratio, which does not exceed 2, can carry out a landing at any point of surface, which stretches to thousands of kilometers in the longitudinal and side directions relative to this trajectory and entrance points in the atmosphere.

Fig. 15 shows maneuverability capabilities of the apparatus, which possesses lift. Dashed curves showed the possible landing

regions, which correspond to different values of lift-drag ratio (number in the curves). Each curve limits the possible landing region of apparatus with the constant lift-drag ratio. The regions between the curves correspond to the possible locations of the landing place of apparatuses with the variable value of quality. Unbroken curve showed the trajectory of the typical maneuver of apparatus with the value of lift-drag ratio of 1.5.

Prior to entry into the atmosphere of planet motion of spacecraft in inactive leg obeys the law of celestial mechanics.

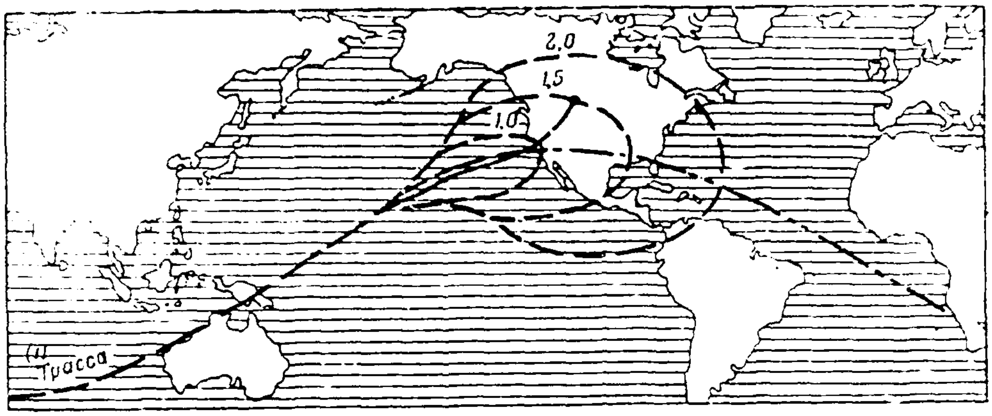

Fig. 15. Maneuverability capabilities of ship, which possesses lift-drag ratio (lift).

Key: (1). Route.

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This means that the ship moves under the action only of inertial and gravitational forces. Aerodynamic forces come into action upon entry into the atmosphere aboard the ship. Aerodynamic resisting force acts opposite to the direction of the speed of apparatus. Aerodynamic lift and centrifugal forces they act perpendicular to the motion of ship. Attracting force always is directed toward the center of planet (Fig. 16).

Dynamics of motion of ship in section entry into the atmosphere is determined by its inherent inertia and resulting of forces enumerated above. Aerodynamic resisting force decreases the speed of the ship, and centrifugal and lifts impart to it acceleration in a direction perpendicular to its motion. Aerodynamic forces, just as

the accelerations caused by them, vary directly to atmospheric density and to square of the speed of ship. In proportion to approach to a planet the ship enters into layers of the atmosphere with the very low density. During further "immersion" in the atmosphere its density rapidly grows/rises, and as a result of the drag the speed of ship begins to decrease.

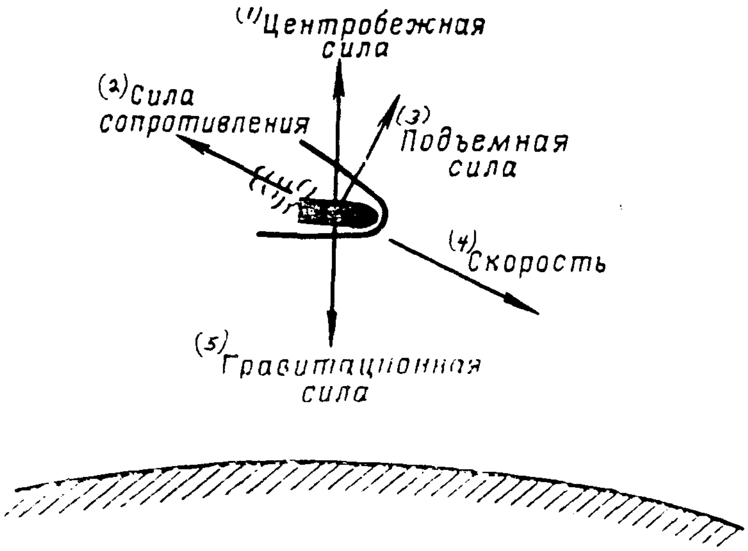

Fig. 16. forces which affect aboard ship with atmospheric descent.
Key: (1). Centrifugal force. (2). Resisting force. (3). Lift. (4). Speed. (5). Gravitational force.

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Thus, the g-force is proportional to the product of two values, one of which increases, and the other decreases. At certain point in the trajectory a decrease in the velocity of ship begins to predominate above an increase in the air density. This leads to the fact that the g-force reaches maximum value, after which begins to decrease.

It should be pointed out that upon return from outer space (in contrast to deorbit of artificial satellite) serious problem will guarantee of precision/accuracy of control, which makes it possible run predetermined program of descent, in particular, carry out descent along most advantageous trajectory and avoid in this case excessively

high g-forces and aerodynamic heating. Flight along the geocentric orbit does not require high precision of the guidance upon entry into the atmosphere, since it is easy to correct too steep an entry by the short-term application of thrust, but with too flat an entry it is possible again to exert a retro impulse. However, in the atmosphere with the speed, which exceeds the first space, the vectoring errors are very dangerous upon entry, since in this case the excessively steep entry can lead to the destruction of ship with the descent, and too flat - to the irretrievable drift of ship into outer space.

Thus, must be certain corridor of entry of the ship in the atmosphere (Fig. 17). The regions which correspond to excessively flat and excessively steep entry in Fig. 17 are shaded. If as a result of vectoring error the ship leaves beyond the lower boundary of the corridor of the entry, then in the atmosphere it will enter at an inadmissibly large angle, undergoing in this case the action of too heavy overloads. But if vectoring error leads to the fact that the ship will leave beyond the upper boundary of the corridor, then it will not be able to enter in the sufficiently dense the layer of the atmosphere and to extinguish the speed with the single "immersion" in the atmosphere. In this case the descent of ship will be produced with the repeated "immersion" in the atmosphere (Fig. 18).

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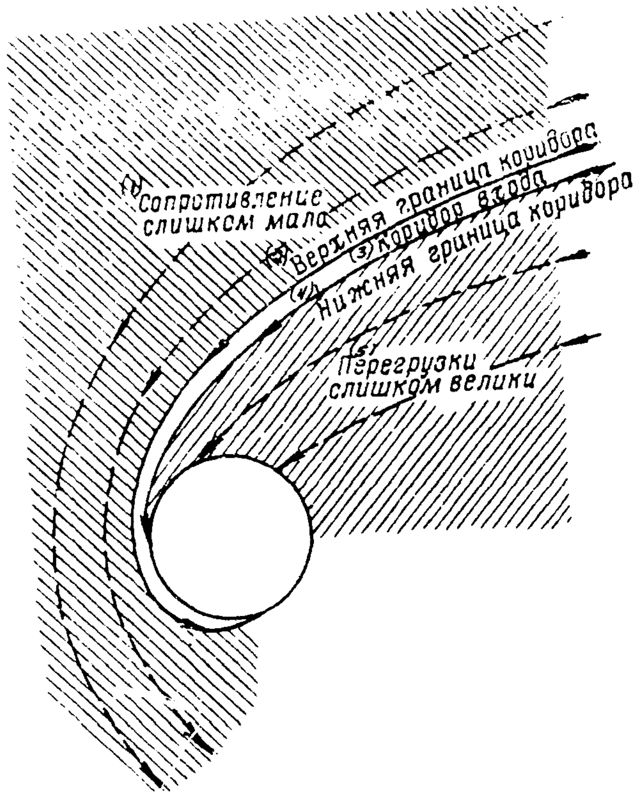

Fig. 17. Corridor of entry of ship in the atmosphere.

Key: (1). Resistance is too low. (2). Upper boundary of the corridor. (3). Corridor of entry. (4). Lower boundary of the corridor. (5). G-forces are too great.

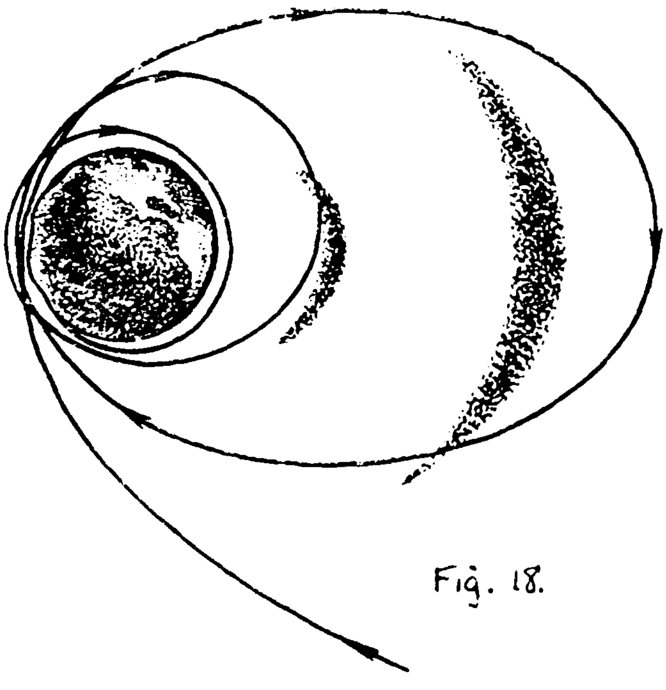

Fig. 18.

Fig. 18. Descent of ship with repeated "immersion" in the atmosphere of Earth.

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As it is reported in the foreign press, until recently the entry into the Earth's atmosphere on the upper boundary of the corridor considered a good resolution of the problem of return from the Moon or of the more remote space expeditions, since the speed of ship decreases with each passage through the atmosphere (during each "immersion" into it). This method of entry would make it possible to solve the problem of aerodynamic heating in the stages: the heat, absorbed by ship with each passage through the atmosphere, will be emitted into outer space in the phase of elliptical trajectory distant from the planet. However, with the discovery of radiation belts this method lost its attractiveness for the manned flights, since the multiple traversal through the zones of radiation can lead to the high radiation doses of crew.

Determination of the width of the corridor of entry into dense layers of the atmosphere is one of the fundamental questions which appear with resolution of the problem of the return of the spacecraft to the earth. Cosmonauts must have available precise information about the angle of entry into the atmosphere, about the speed and

flight altitude upon the entry, which to virtually ensure is very difficult. Therefore usefuliy in more detail to examine, to what extent will change the width of the corridor, caused by errors in the values indicated.

It is established that significant part of corridor near its upper boundary in practice cannot be utilized due to increased sensitivity of trajectory to errors along angle of entry. However, the maximum angular velocities and the accelerations necessary for the flight in limits of the corridor are comparatively small and are 10 deg/s and 1 deg/s^2, respectively.

Corridor of entry is sufficiently wide for low speeds and errors in navigational information on the whole do not create special difficulties. At the rates of entry, which exceed 15000 m/s, the corridor becomes so narrow that also the low errors in the determination of the angle of entry, altitude and speed can lead to the great difficulties in the process of descent, even if control system in the middle trajectory phase derives ship into the prescribed/assigned corridor.

The corridor of the entry, schematically depicted in Fig. 19, is determined by two trajectories, which are its boundaries.

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As upper boundary serves the flat trajectory, moving along which,

apparatus still is held in the atmosphere, i.e., it does not emerge from the dense layers of the atmosphere at a velocity, which exceeds circular. As lower boundary serves the steep trajectory, during the motion along which the aerodynamic g-force, which affects aboard the spacecraft, does not exceed the permissible value.

Let us accept maximum permissible g-force of equal to 10. The corridor of entry is usually quantitatively rated/estimated by altitude difference of the fictitious perigee of these two limiting trajectories. To sometimes more conveniently determine the corridor by the angles of the slope of the corresponding trajectories at the fixed/recorded height/altitude. This height/altitude can be accepted equal to 120 km, since in the investigations, dedicated to descent in the atmosphere, usually this value of height/altitude corresponds to the upper boundary of a sufficiently dense atmosphere.

For us only the trajectory phase, included between the entry point into the atmosphere at an altitude of 120 km and the point where the angle of the inclination-trajectory becomes equal to zero, is of interest. We will call the latter the exit point from the dive.

One should emphasize that not entire corridor of entry included between the upper and lower boundaries can prove to be suitable for descent. Let us examine at first the apparatus, which maneuvers along the bank.

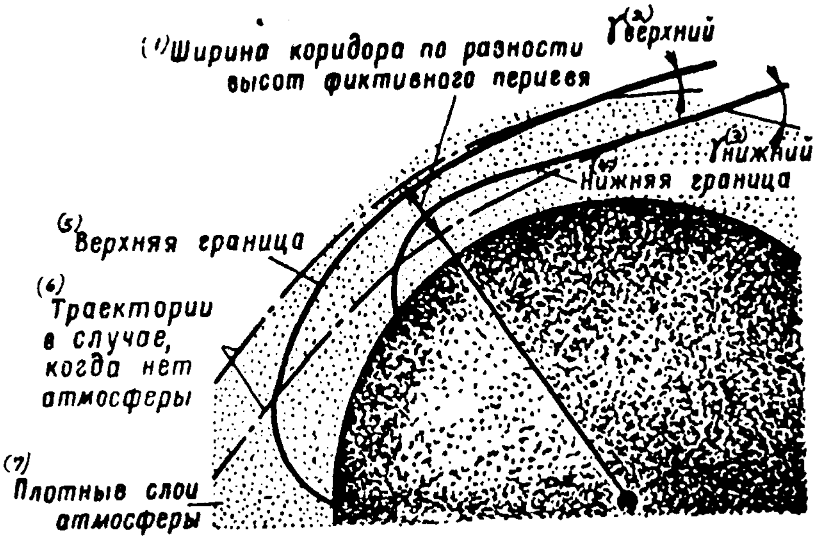

Fig. 19. Determination of the corridor of entry into the atmosphere. Key: (1). Width of the corridor on altitude difference of fictitious perigee. (2). Upper. (3). Lower. (4). Lower boundary. (5). Upper boundary. (6). Trajectories in the case when there is no atmosphere. (7). Dense layers of the atmosphere.

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In particular, as it is indicated in the foreign press, a change in the roll attitude is utilized in the diagram of control of an apparatus of the type "Apollo". The angle of attack of this apparatus in the process of descent remains constant, and the control of motion according to trajectory is accomplished via roll turns, which change the direction of the vector of lift. For simplicity, the entry into the atmosphere with the fixed attitude of roll, which remains constant, is accepted to the exit point from the dive. For such apparatuses a certain corridor of entry with the specific upper and

lower boundaries corresponds to each fixed value of roll attitude. A change of the boundaries of the corridor in the dependence on the roll attitude is shown in Fig. 20 for the rates of entry of 15200 and 19800 m/s.

The roll attitude, plotted along horizontal axis, is counted off in such a way that its zero value corresponds to horizontal position of apparatus, when lift is directed accurately upward, and value of 180° corresponds to inverted flight position of the apparatus when the lift is directed accurately downward. Along the vertical axis the angle of entry into the atmosphere, which is the flight path angle at the calculation point at the altitude of 120 km, is plotted.

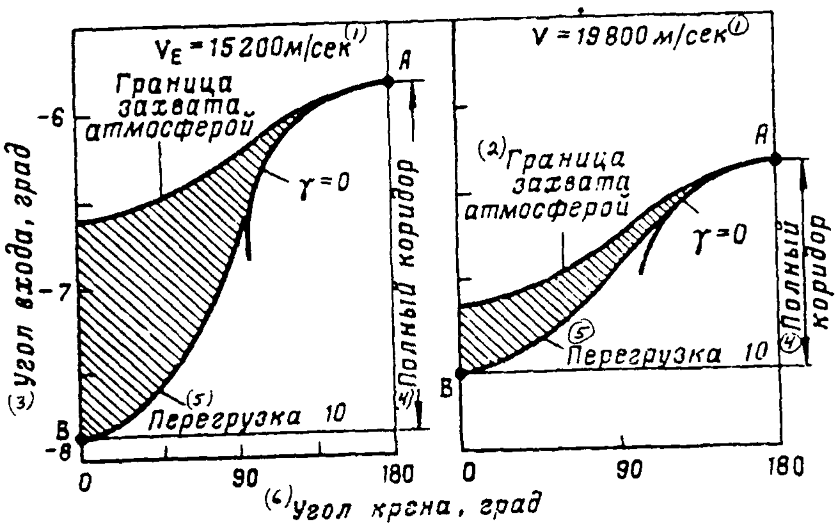

Fig. 20. Flight conditions of apparatuses entering in the atmosphere with a fixed attitude of roll (lift-drag ratio to equal 1; velocity head q=488 kgf/m²).

Key: (1). ... m/s. (2). Capture boundary by atmosphere. (3). Angle of entry, deg. (4). Complete corridor. (5). G-force. (6). Attitude of roll, deg.

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The shaded region corresponds to all possible values of angles of entry and roll, at which there occurs the "capture" of apparatus by the atmosphere without the excess of the g-force, equal to 10. In the region, where the lift is positive, i.e., in the range of the attitudes of roll of 0-90°, the lower boundary of the corridor is determined by the permissible limit with the g-force, equal to 10.

At large attitudes of roll, when vectored lift is directed down,

lower boundary is determined by zero value of slope angle. The value of the difference between the angle of entry, which corresponds to the upper limiting trajectory of descent with the lift, directed down (point A in Fig. 20), and the angle of entry, which corresponds to the lower limiting trajectory of descent with the lift, directed upward (point B), characterizes the corridor approximately, by its one measurement. In actuality, the corridor is determined by two values - angle of entry and roll attitude. At rates of entry less, for example, 15000 m/s, one characteristics the corridor, apparently, it is sufficient. However, at greater speeds it is important to consider the interconnection between the angle of entry and the orientation of the apparatus along the bank. This means that at this angle of entry, which lies within limits of the corridor of the entry, for guaranteeing the design conditions of descent it is necessary that the range of a change in the roll attitude would lie/rest at the shaded region. Hence, obviously, emerges requirement about the need for having sufficiently precise information about the angle of entry, so that the selected roll attitude would correspond to the permissible region of its change.

Let us produce evaluation of apparatus, entering in the atmosphere with a velocity of 19800 m/s at an angle of -7°. Under the given conditions for guaranteeing the "capture" of the apparatus with the atmosphere its roll attitude must range from 50 to 85°. If in the flat limiting trajectories the angle of entry is determined by control system with an accuracy to 1/4°, then the permissible range of roll

attitudes will compose already 95-110°. It is evident from Fig. 20 that with any values of bank in the limits of this range (for the angle of entry of -7°) the maximum g-force will exceed 10. Similarly error in 1/4° in the steep limiting trajectories will lead to the "nonseizure" of apparatus by the atmosphere.

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It is evident from Fig. 20 that if the shaded field is narrow, then there is necessary the high precision of the determination of the angle of entry, so that it would be possible to select this attitude of roll, which would provide successful descent. Width of shaded band is, thus, by the modulus of precision, for which aboard must be known the angle of entry.

In connection with this, it is very important to estimate the effect of different parameters on width of region indicated, and also, therefore, on width of useful part of the corridor of entry. Flight conditions can be characterized by two parameters - with the width of the complete corridor and with the maximum width of the shaded region of the permissible flight conditions, by which nonrotating the apparatus reaches with the value of the attitude of roll of 0°. Both these characteristic widths of the corridor decrease with an increase in the velocity. This it confirms Fig. 21, in which is shown the dependence of the width of the complete corridor and width of the corridor of entry with the zero angle of bank on the speed. The dependence indicated characterizes the requirements, presented to the

precision/accuracy of control at different stages of space flight.

According to reports to foreign press, at entry velocity of 10970 m/s the difference between the two curves indicated will be not especially essential, since the width of corridor of entry with zero angle of bank composes more than 80° widths of complete corridor.

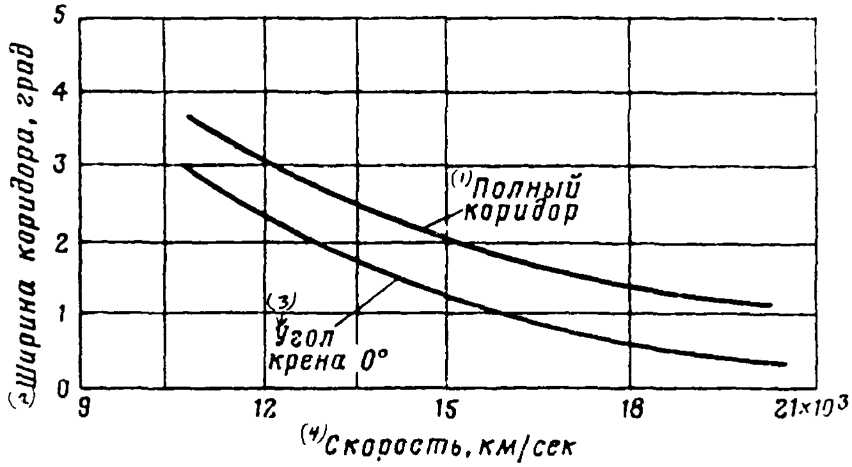

Fig. 21. Effect of the rate of entry on width of corridor (lift-drag ratio to equal 1; velocity head q=488 kgf/m²).

Key: (1). Complete corridor. (2). Width of corridor, deg. (3). Roll attitude. (4). Speed, km/s.

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However, at the rates of entry which exceed 15000 m/s and corresponding to return from the interplanetary flights/passages, the corridor of entry with the zero angle of bank is only the low part of the complete corridor.

Thus, at high speeds system of flight control in deep space must possess approximately three times larger precision/accuracy than system, intended for flights to Moon.

On the width of the corridor of entry the value of the lift-drag ratio also has an effect. This can be seen well from Fig. 22. The

width of the complete corridor grows with an increase in the lift-drag ratio until it achieves the value, close to one. Further increase of quality leads to a less noticeable increase in the width of the corridor.

Since requirements, presented to precision/accuracy of control of apparatuses with fixed/recorded angular orientation according to pitch and bank, are very high, let us rate/estimate possibilities of aerodynamic maneuver with descent in the atmosphere.

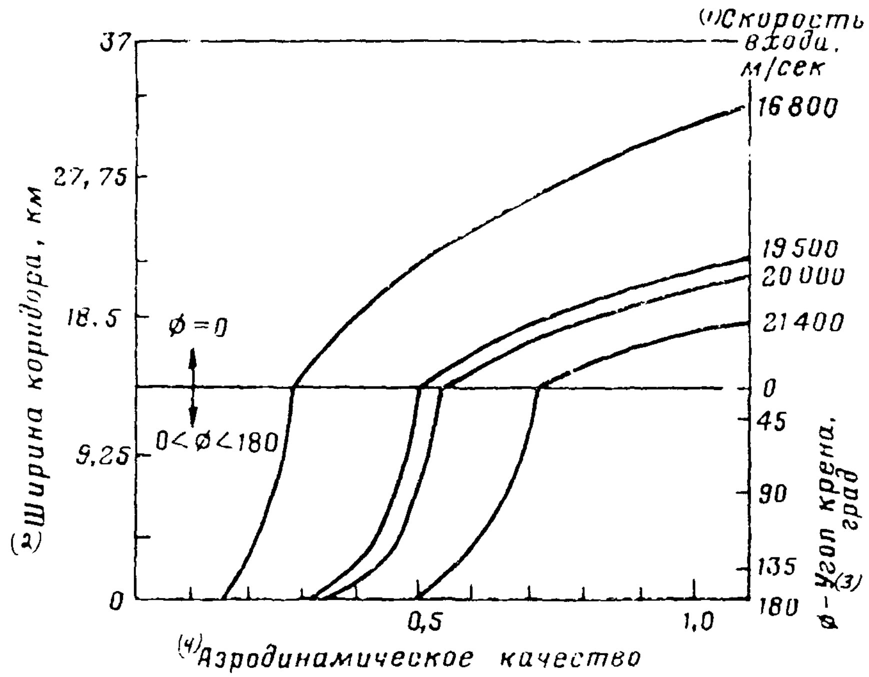

Fig. 22. Effect of value of lift-drag ratio on width of corridor of entry in the atmosphere.

Key: (1). Rate of entry, m/s. (2). Width of corridor, km. (3). Attitude of roll, deg. (4). Lift-drag ratio.

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The realization of it is possible only during the sufficiently deep "immersion" of apparatus in the atmosphere, when, in the first place, velocity head is so great that the aerodynamic forces substantially affect the motion of apparatus, and, in the second place, the entering the onboard system for control information about the g-force makes it possible to select the correct program of the maneuver. In connection with this it is desirable to select the smaller value of the calculated fixed/recorded height/altitude of entry so that point in

the trajectory corresponding to it could serve as the beginning of aerodynamic maneuvering.

The accomplishing of the aerodynamic maneuver with descent in the atmosphere is connected with need for control of vectored lift, which can be achieved either by rotation of the apparatus along bank, which changes the direction of the vector of lift, or by pitch control. It goes without saying, is possible also the combination of these methods.

Investigations show that, since atmospheric density at the height which corresponds to initial section of descent, independent of conditions for entry is too low for maintaining constant g-force, equal to 10, lift coefficient of apparatus, which satisfies maneuver on pitch, first must be equal to its upper limiting value. When apparatus more deeply enters in the atmosphere and velocity head will become sufficiently large for maintaining the prescribed/assigned tenfold g-force, it is necessary to continuously decrease the lift coefficient, and simultaneously with it, the drag coefficient. During further motion along the trajectory, the maximum value of velocity head is attained.

It must be noted that if system of flight control in deep space accurately derives/concludes apparatus into corridor of the entry, then for the control of pitching it is not required that onboard of descent control system would have available any other information

about conditions for entry into the atmosphere, except continuous information about g-force. It is possible to obtain it directly from the onboard accelerometers. Then it is possible to continuously change controlling moment/torque in order to support any prescribed/assigned size of the g-force, without having data about the flight path angle, about the speed or altitude.

From the aforesaid it is possible to draw this conclusion: the greater the available range of change in lift coefficient has the space vehicle, the wider its corridor of entry in the atmosphere.

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LANDING OF SPACECRAFT.

At present the landing on earth of all returning space vehicles is accomplished with the aid of parachutes - relative to simple, reliable and proven devices. Subsequently, they will also utilize them for landing the space vehicles. However, since the standard parachutes have actually vertical and unguided descent, then they will be used increasingly less and less. Besides the usual parachute, there can be used the steerable parachute, which ensures the possibility of the selection of the landing place. Fig. 23 shows the steerable parachute, whose tests were conducted in the USA. The flap is its special feature. It is the part of the canopy located at the edge and the having the possibility to diverge upward. However, this flap provides only a limited possibility of the selection of glide

path.

A flexible wing is another device for the rescue of the flight vehicle. The fundamental elements of its construction/design are an inflatable fin and wing edges. General idea about of landing system with this wing gives Fig. 24. Flexible wing increases the ability of apparatus to plan, which has high value from the point of view of the reaching of a safe landing place.

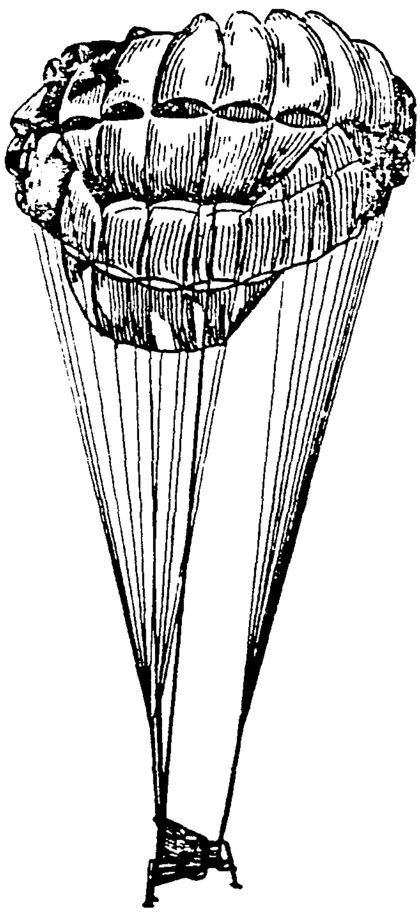

Fig. 23. Descent of a spacecraft by a maneuvering parachute.

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But, creating greater than parachute, planning ability, flexible wing is at the same time a more complex system and, probably, for it there will appear more the problems of aerodynamic character than for the parachute.

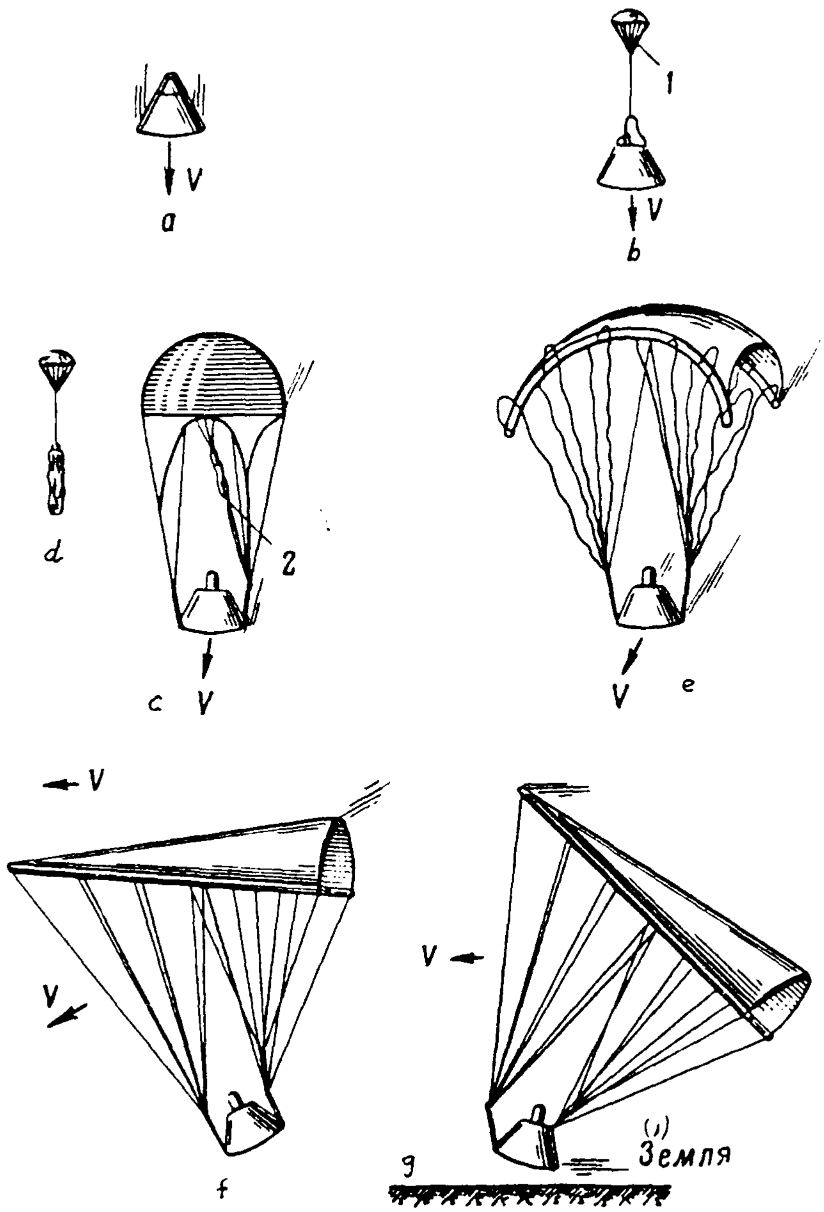

Fig. 24. Stages of landing spacecraft with the aid of a flexible wing: 1 - brake parachute; 2 - flexible anti-flutter elements.

Key: (1). Earth.

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Maximum value of lift-drag ratio for flexible wing with thick inflatable leading edges was obtained equal to 3.5 at an angle of attack of approximately 40°. It is possible to obtain for a wing with thinner leading edges and with the great lengthening, as they assume, the larger the value of lift-drag ratio.

Touchdown of spacecraft with use of flexible wing occurs thus. After braking of ship with the ballistic descent in the dense layers of the atmosphere to the small supersonic or high subsonic speeds (Fig. 22a) is opened brake parachute 1 (b). It extracts the flexible wing (c), packed in the sack within the hull of ship, after which the parachute and sack are separated from the wing (d). Then the fin and wing edges is inflated by gas (e). The ship begins to be lowered on the flexible wing (f) with a small (to 6 m/s) rate of descent prior to the moment of touchdown (g).

Control of a flexible wing can be accomplished either aerodynamically or by displacement/movement of the center of gravity of the system relative to wing via different aspect ratio and shortening of front/leading and rear cord of wing (control according to pitch angle) and cord, that go to tips of wing (roll control).

Conducted investigations showed that success of application of flexible wing in many respects depends on correctness of its disclosure/expansion. The least error in the coordination of operations on the time can seriously hinder/hamper stabilization and

alancing/trimming of system and consequently, prevent a safe landing.

One of American firms develops/processes system of return of hip, which must utilize flexible wing, produced after braking of apparatus in ballistic trajectory phase to subsonic speed at heights/altitudes on the order of 15000 m. This design (Fig. 25) is a wing with an aspect ratio of 2.3 and sweepback on leading edge of 55°. ts framework/body is formed by inflatable "beams" - by wing edges and in, which goes along the center line of wing. The diameter of the eams" comprises 6% of the chord length. Spacecraft is fastened to exible wing five with cords.

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Three of them are conducted/supplied to the keel and on one - to each wing edge. In this construction/design a change in the length of lateral or keel cord also makes it possible to control/guide flexible wing along the pitch and the bank. Maneuvering along the bank (lateral maneuvering) provides great possibilities in the selection of landing field to the spacecraft crew.

As show experiments wind tunnels, flexible wing possesses sufficiently high aerodynamic characteristics. Some specialists confirm that it will provide the lifting efficiencies qualitatively compared with the appropriate characteristics of the rigid wing of equal aspect ratio.

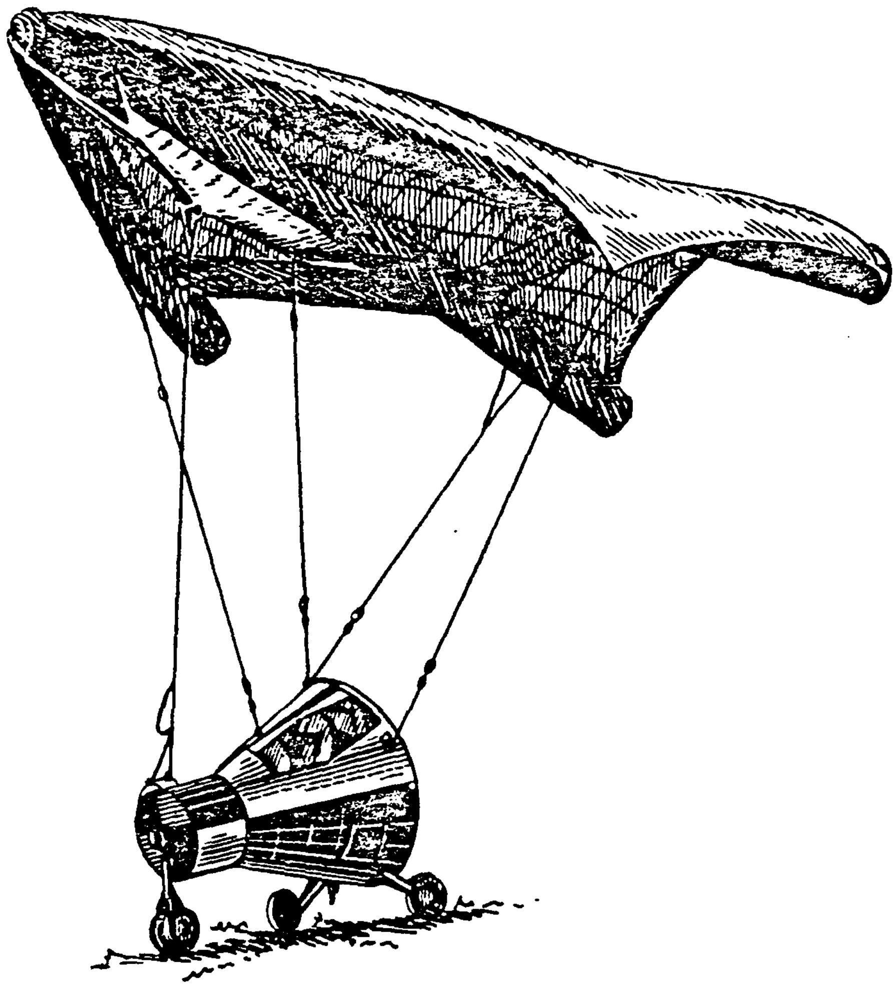

Fig. 25. Flexible wing.

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With the production of the flexible wing at an altitude of 12 km the range of gliding flight can reach from 36 km with a lift-drag ratio of 3 to 48 km with a lift-drag ratio of 4. Over the long term it is proposed to bring the range of gliding flight to 70-90 km.

In press/printing it is indicated that this wing, equipped with

automatic of landing system, can have approach speed about 140 km/h, and rate of fall to 55-65 km/h at a vertical velocity from 0 to 2.5 m/s.

In the course of tests of a flexible wing, the shortcomings connected with its disclosure/expansion and stabilization in flight were revealed, which somewhat retarded the adjustment of landing system.

Were carried out two flights with pilots aboard. In one flight the pilot was was forced to be catapulted due to the poor controllability, the secondly - during the rough landing of the mock-up of spacecraft the pilot was injured.

Subsequently, three additional copies of the flexible wing were tested. One of them was controlled by radio, and pilots were located on board two of others. In all were carried out 12 flights. Flexible wing was towed by helicopter. At the height/altitude of 3000 m it was disengaged and passed into the free flight, which continued for 4.5-5 min. The pilots confidently made a landing within the limits of 150 m from the center of a 30-meter circle. Successfully also was finished testing of a flexible wing with its control on the radio.

By the way, it is proposed to utilize flexible wing for landing loads from transport aircraft, since with its aid it is possible to accomplish/realize more precise, than with the aid of parachutes,

landing on small open areas located at a sufficientl
from the place of separation of loads from the aircra
construction/design this wing, as they note, will b
the wing, used for landing the spacecraft. With it
height/altitude of 9000 m the gliding distance is ap

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Experiments established that it possesses good maneu
sufficiently rapidly changes the direction of flight
commands/crews from the Earth or automatically and pr
precision/accuracy of touchdown.

For landing spacecraft is developed/processed also system with rotor. Upon the atmospheric entry the blades/vanes of rotor are in the folded state, during the landing they straighten and is created lift. The ship, which has a landing system with a rotor, can plan, descend vertically, accomplish a landing in an assigned place with the horizontal and vertical velocities close to zero. From this point of view this system attracts considerable attention of the specialists.

The experience of conducting landings of helicopters with engines off accumulated at the present time shows that landing apparatuses in prescribed/assigned place with rotor with a load of up to 20 kgf/m² is accomplished/realized sufficiently easily. (To 80 kgf/m² without the special automation a landing to carry out proved to be quite difficultly with a larger load.

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Chapter 2.

ENGINES OF SPACE VEHICLES AND ONBOARD ENERGY SOURCES.

In resolution to the problem of space flights a fundamental factor is energy. Energy is necessary for the injection of spacecraft into orbit, of its transition/transfer from one orbit to another, the completion of flight to other planets, guaranteeing landing ship to the earth and on other planets, etc. Energy is necessary also for the guarantee to the crew of the necessary living conditions in space, for the operation of equipment of the means of communication and electro-instrument equipment. However, prolonged flights to such planets distant from the earth as Mars, or even those more distant from it will require the continuous work of apparatus, equipment and maintenance of vital conditions for the crew in the unusual situation for several years. In connection with this the problem of the resource/lifetime of the energy sources for the interplanetary spacecraft acquires most important importance.

Largest quantity of energy is utilized to that in order to launch ship from Earth into outer space. During the starting/launching of the space ship "Vostok", in which was situated Yu. A. Gagarin, the power of rocket engines composed approximately 20 million horsepower.

For flights with landing on other planets it is necessary to have

a thrust of the power plant capable of overcoming gravitational forces on surface of planet.

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Sufficient for this is considered the thrust whose ratio to the weight of spacecraft is more than one. One should, however, note that the value of required thrust on the Moon and on Mars will be less than on the Earth, since on these planets is less than on the Earth, the value of gravity force.

ROCKET CHEMICAL-PROPELLANT ENGINES.

At present space technology rests in essence on rocket engines, which operate on chemical fuel/propellant. As a result of the combustion of this fuel/propellant very high energy during the small time interval is developed.

The fuel/propellant can be liquid and solid. The type of the fuel/propellant used determines the name of this engine. If the fuel/propellant is liquid, the engine is called a liquid propellant rocket engine (ZhRD), if solid - by solid-propellant rocket engine (RDTT).

Principle of operation of the ZhRD lies in the fact that into its combustion chamber from tanks located on the rocket, under pressure the fuel and oxidizer is supplied. As a result of their combustion is

formed a large quantity of gases, which enter jet nozzle and at a high speed escape from it. In this case the reactive thrust is formed (Fig. 26).

In an RDTT the propellant, which is found in a solid state, is placed directly in combustion chamber (Fig. 27). It unites in itself both the fuel and oxidizer. During the burning of propellant/fuel charge are formed the combustion products, which emerge at a high speed from engine nozzle, creating in this case reactive thrust.

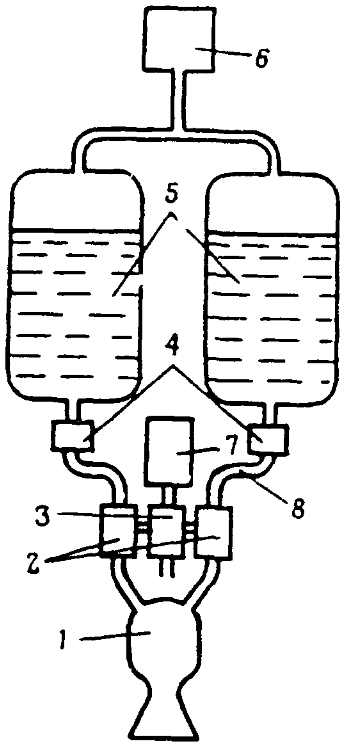

Fig. 26. Schematic diagram of liquid propellant rocket engine: 1 - combustion chamber; 2 - pumps; 3 - turbine; 4 - valves; 5 - fuel tanks; 6 - device which feeds compressed gas into tanks; 7 - gas generator; 8 - pipleline for supply of fuel/propellant.

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Rocket engines which operate on solid fuel are convenient in the operation but have lower specific impulse than the ZhRD.

From perfection of rocket engines in the final analysis depends payload weight, concluded in orbit or into outer space

Soviet scientists and designers created powerful liquid

propellant rocket engines. One of them is engine RD-107, developed in 1954-1957. It works on the high-energy propellant - liquid oxygen and the hydrocarbon fuel.

Engines RD-107 and their modifications are utilized as power plants of first stage of a series of carrier rockets. They ensured successful injections into orbit of the manned space ships "Vostok" and "Voskhod", and also the successful start of a number of other space vehicles.

According to its characteristics the engine RD-107 exceeds world models of ZhRD on the same fuel/propellant of identical period of creation. Its thrust in the void is 102 T.

Device/equipment of engine RD-107 is shown in Fig. 28. Engine has four fundamental 5 and two steering 1 combustion chambers. Fuel and oxidizer are supplied into the combustion chambers with the aid of a turbopump unit (TNA).

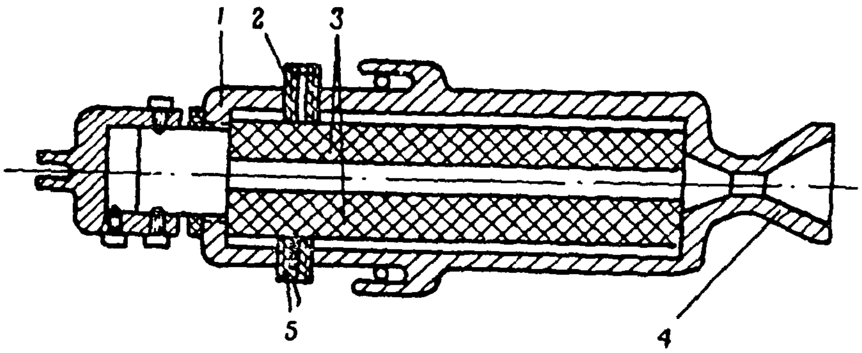

Fig. 27. Schematic diagram of a solid-propellant rocket engine: 1 - housing; 2 - safety valve; 3 - solid-propellant grain; 4 - nozzle; 5 - device for burning.

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Into the design of the turbopump unit enter two fundamental centrifugal pumps (fuel and oxidizer) and two supplementary pumps (hydrogen peroxide for the feed/supply of gas generator 7 and liquid nitrogen for the feed/supply of the pressurized system of the fuel tanks and oxidizer by gaseous nitrogen).

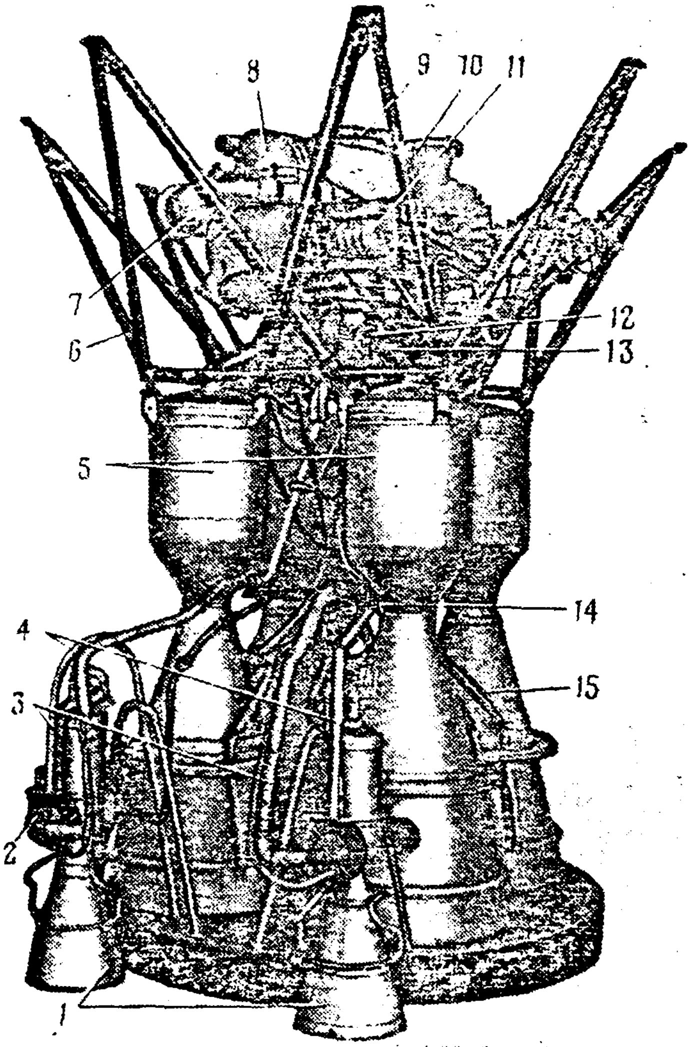

Fig. 28. Liquid propellant rocket engine RD-107: 1 - steering combustion chambers; 2 - unit of fluctuation and supply of oxidizer; 3 - oxidizer pipes of steering chambers; 4 - brackets simulated (in construction/design they are absent); 5 - main chambers; 6 - power frame; 7 - gas generator; 8 - housing of heat exchanger on turbine; 9 - intake pipe of pump of oxidizer; 10 - intake pipe of fuel pump; 11 - pressure sensor in combustion chamber; 12 - main valve of oxidizer; 13 - oxidizer pipes; 14 - main valve of fuel; 15 - fuel pipes.

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In the gaseous state nitrogen is converted in tubular heat exchanger 8, warmed by the exhaust gas.

With operation of the engine hydrogen peroxide enters gas generator, where steam gas is formed. Steam gas enters the turbine of turbopump unit and it puts it into the action. In this case the pumps supply propellant components through a large quantity of injectors into the combustion chamber, where their inflammation occurs. Gases emerge at a high speed from the engine. As a result of the outflow of gases reactive thrust appears. The engine thrust is regulated by changing the expenditure/consumption of steam gas, which enters from the gas generator.

For cooling the internal combustion chamber walls, heated in process of combustion of fuel/propellant to high temperature, fuel can be used. Before the entrance into the injectors it passes through the extra-jacket spaces of combustion chambers, cooling internal walls outside. Furthermore, the fuel, which is fed through the peripheral injectors of combustion chambers, forms the peculiar curtain of cooling.

The launching, control of operation of engine and engine cutoff are accomplished/realized automatically on commands/crews from onboard of rocket.

The power plant of the second stage of rocket "Vostok" was the ZhRD, basically analogous in design to engine RD-107. Its difference is in the fact that it has four steering combustion chambers, and not two as in RD-107. There are also some peculiarities associated with more continuous firing, since it is started with the missile takeoff simultaneously with the first-stage engines.

Another representative of Soviet liquid propellant rocket engines is engine RD-119 (Fig. 29). Its thrust is equal to 11 T. Engine is designed in 1958-1962. Liquid oxygen (oxidizer) and unsymmetrical dimethylhydrazine is fuel/propellant (combustible). Installed is the engine on the second stage of rocket "Kosmos".

Modern construction/design of engine provides its high reliability in operation.

Soviet rocket engine construction is being continuously developed.

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Utilizing the newest achievements of science and technology, and also acquired experiment, our scientists and designers developed the powerful small/miniature rocket engines of the most modern diagram with the very high energy characteristics.

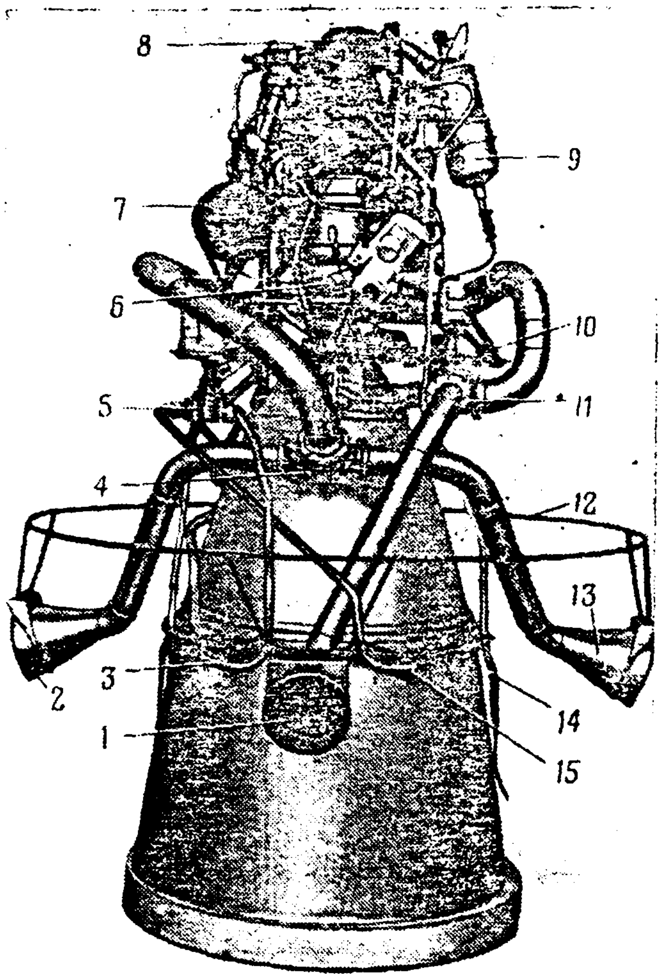

Fig. 29. Liquid propellant rocket engine RD-119: 1 - steering nozzles of pitch (second nozzle from opposite side); 2, 13 - steering nozzles of yaw; 3, 15 - steering nozzles of bank (second pair of nozzles from opposite side); 4, 5, 11 - gas distributors with electric drives; 6 - combustion chamber; 7 - spherical air bottle; 8 - turbopump unit; 9 - gas generator; 10 - power frame; 12 - mounting ring of steering system (in construction/design of engine it is absent); 14 - removable cap.

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These engines are installed on the carrier rocket "Proton" created in 1965. The total maximum net power of the engine installations of this rocket is triply more than rocket "Vostok".

Rocket-space system "Proton" offers new possibilities in injection into orbits of complicated and large in weight space objects.

COMBINED VRD FOR SPACE VEHICLES.

For the launching of space vehicles, it is expedient to use jet engines. Entire the fact is that in this case it is possible to utilize as one of the components of propellant (oxidizer) - air, which gives serious gain in the efficiency/cost-effectiveness. However, in liquid-propellant rocket engines it is necessary to reserve oxidizer on board the rocket. But this increases its weight, which in turn decreases payload weight, concluded in orbit.

For first stage of booster aircraft it is possible to utilize usual turbojet engine (TRD [turbojet engine]), which will accelerate/disperse apparatus in dense layers of atmosphere.

In TRD the air taken from the environment by an air intake is compressed by a compressor, and then is supplied into the combustion chambers, where it is mixed with hydrocarbon fuel and burns. The

resultant combustion products are expanded in the turbine and in the nozzle. Turbine sets in motion compressor, and during the expansion of working medium/propellant in the nozzle to flight vehicle is communicated thrust.

Turbojet engine has high effective value of specific impulse. In the newest engines it can reach ~1500—2000 s, whereas in the ZhRD on a chemical fuel/propellant it is equal to ~300— 400 s. Furthermore, this engine has the high value of thrust-to-weight ratio. This combination of data of the TRD is very attractive from the point of view of the use of these engines for the starting/launching of space vehicles, including piloted.

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If we replace the first stage of the rockets with ZhRD or RDTT by an accelerator (booster), on which several turbojet engines are installed, and which is capable after the accomplishment of its objective of returning to the earth (Fig. 30), then it will be possible to obtain a cheaper means of removal of large payloads in orbits of satellites or in trajectory of spacecraft. The booster aircraft can be used as this accelerator. At the altitude approximately 20-25 km booster aircraft is separated/liberated from the space vehicle and in the action enters the second rocket step/stage, which works on the chemical or nuclear fuel, and and possible being hypersonic jet engine.

One should, however, indicate series/row of limitations, characteristic to turbojet engine. It gives economic usefulness only with the possibility of its repeated use.

During single use TRD already becomes economically unfavorable its application. Cost of the TRD considerably exceeds the cost of the contemporary rocket engine.

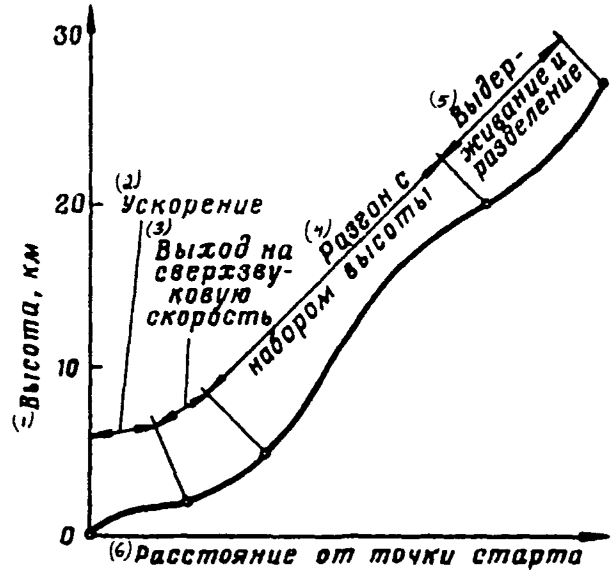

Fig. 30. Standard trajectory of accelerator (booster), on which there are installed TRD.

Key: (1). Altitude, km. (2). Acceleration. (3). Output to supersonic speed. (4). Acceleration with climb. (5). Maintaining and separation. (6). Distance from launching point.

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Turbojet engine of usual diagram can ensure operation up to the cruising speed, which corresponds to number $M=3{,}5\div4{,}5$. Higher than these speeds it is not applied. Why? The TRD is advantageous in this speed range of the flight, in which the total pressure of gas behind the turbine is higher than the complete air pressure at the entry into the compressor. But if pressures are equal, then turbocompressor becomes unnecessary. In this case it is possible from the air intake to directly direct air into the combustion chamber of engine. Equal

pressures are reached exactly at the flight speeds, which correspond to number $M=3{,}5\div 4{,}5$. This means that for the march hypersonic flight it is expedient to use the ramjet engines (PVRD [ramjet engine]). But this engine, being highly efficient at the hypersonic cruising flight speeds, cannot independently start and be accelerated to cruising speed and does not possess a good efficiency at the subsonic and low supersonic flight speeds. However, the turbojet engine, on the contrary, has good data on acceleration, subsonic and supersonic flight speeds.

The thought to combine engines of both types into one power plant arose. This can be made either by a combination of the engines of two types or by their organic unification in one unit. The engine, in which organically are united turbojet and ramjet engines, was called direct-flow turbine. This engine provides prolonged sustained flight of the manned flight vehicle with the hypersonic speeds at the heights/altitudes of 30 km and more; it can be utilized at the second stage of space vehicle. The schematic of the device/equipment of turboramjet engine on the base of turbofan engine is given in Fig. 31. On the takeoff and acceleration to cruising speed it works as turbofan engine with the afterburner. A shutter/valve 1 is installed in the upper position. In this case air from the low-pressure compressor (KND) enters partially into the high-pressure compressor (KVD) - the first duct - and partially along pipe ring 2 into afterburner 3, which is included in this mode of operation of the engine. On the cruising flight speed (M=3-4) shutter/valve 1 is set in lower position. In

this case air enters only into the afterburner and the engine it works as a PVRD.

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On the landing modes shutter/valve 1 is set in upper position. Afterburner is switched off. Engine works as usual bypass engine, providing high efficiency/cost-effectiveness.

As we see, in the turboramjet engine the so-called circuit regulation is used, i.e., due to change in adjustable elements of engine (air intake and exhaust nozzle) it in dependence on requirement works according to different diagrams: as turbofan engine without afterburning as turbofan engine boosting and as a direct-flow/ramjet VRD.

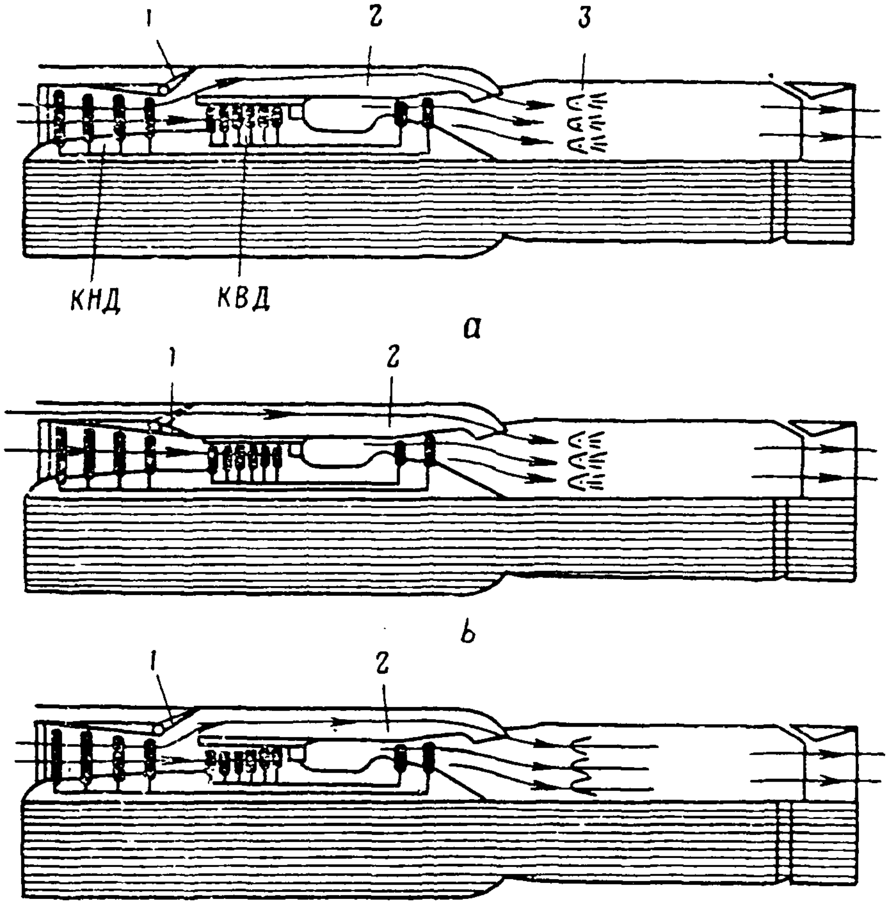

Fig. 31. Diagram of a device of turboramjet engine on the base of turbofan engine: a) takeoff and acceleration, afterburner is included; b) cruise; engine works as ramjet engine; c) landing, afterburner is switched off; 1 - shutter/valve; 2 - pipe ring; 3 - afterburner.

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Obviously, the control system of engines of such type is more complicated than usual.

Besides similar combination of hypersonic engines, there can be and others, for example, TRD and rocket engine. This engine is called turborocket. Fig. 32 gives one of the schematics of turborocket engine. In the combustion chamber 6 the fuel and oxidizer are supplied under the pressure. Here occurs the incomplete combustion of fuel mixture with an increase in the temperature to the allowed values, determined by efficiency of the blades of turbine (827-1023°C). Combustion products enter turbine 3, and then into afterburner 4. Air, compressed in the compressor, is mixed in the afterburner with the gases, which go from the turbine. In the afterburner injects itself the fuel/propellant, which burns out in the atmospheric oxygen. In this case the temperature is raised to 1527-1727°C. In the combustion chamber can be supplied the monopropellant. Turbine rotates relatively low-pressure compressor 2.

Characteristics of turborocket engines on speed and flight altitude and their basic datum (specific weight, specific fuel consumption, frontal thrust) - intermediate between turbojet engines and liquid propellant rocket engines.

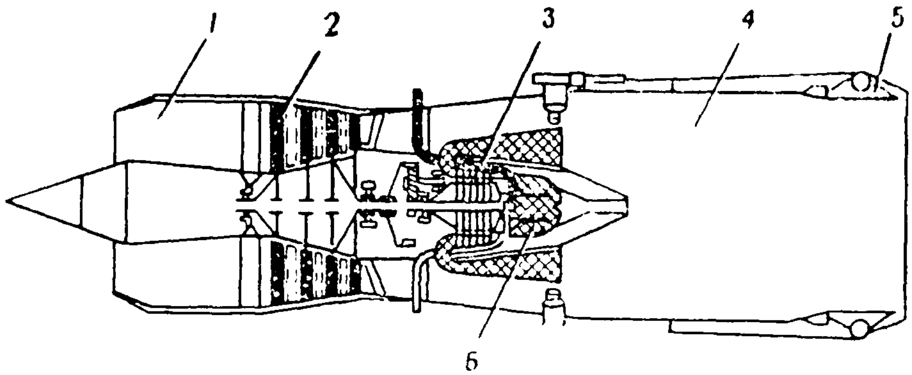

Fig. 32. Diagram of a turborocket engine: 1 - air intake; 2 - compressor; 3 - turbine; 4 - afterburner; 5 - adjustable jet nozzle; 6 - combustion chamber (gas generator).

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In one of the systems, intended for injection into orbit of space vehicles, the use of a turborocket engine in first returned to the earth stage of spacecraft is provided for. This stage is a winged manned booster aircraft.

Let us give the calculated version of this system, intended for injection into orbit of Earth satellite with weight of 230 kgf. Apparatus consists of two stages. The first stage is booster aircraft with two turborocket engines, which work on kerosene and oxygen. Weight of booster aircraft 46 T. The second step/stage - rocket RDTT, placed in the fuselage of booster. Its weight is 12 T.

Booster aircraft is accelerated/dispersed with turborocket engines to speed, which corresponds to number M=4.5. At this speed it

at the altitude of 22500 m is separated/liberated from the rocket and returns to the base.

Let us assume that the speed of space vehicle during its dispersal/acceleration in dense layers of atmosphere must increase in the program given in Fig. 33. From the figure one can see that the speed, which corresponds to number M=4.5, will be achieved at an altitude of 22500 m in 8 min from the moment of takeoff.

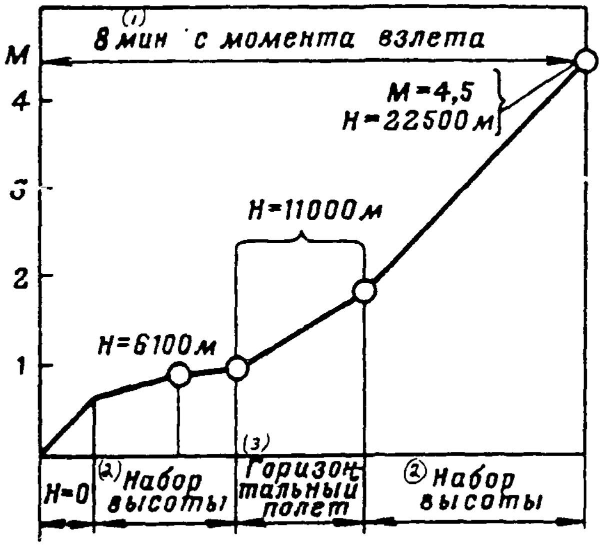

Fig. 33. Flight program of booster aircraft.

Key: (1). 8 minutes from the moment of takeoff. (2). Climb. (3). Level flight.

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Fig. 34 gives weights of engines of different types and fuel/propellant in total weight of aircraft, which require for realization of program of increase in speed and flight altitude accepted. It is evident that the use of a turborocket engine makes it possible to obtain the minimum of the total weight (engine + fuel/propellant), equal to 39% of the takeoff weight of aircraft, whereas for the TRD it is equal to 44%, and for the ZhRD - 58%.

Turborocket engine has smaller weight than the TRD (17% against 24%), and accelerates/disperses aircraft to speed, which corresponds to number M=4.5, for the time twice less (from the moment of takeoff)

than that of TRD. With the minimum total weight relative fuel load on the aircraft with the turborocket engine is equal to 22%, and with TRD - 20%.

Possibility of rapid acceleration of the aircraft and light turborocket engine make this system of very attractive.

rocket of engine, designed for speed, which corresponds to number M=4.5, has smaller number of compressor stages and a smaller compression ratio than that of TRD.

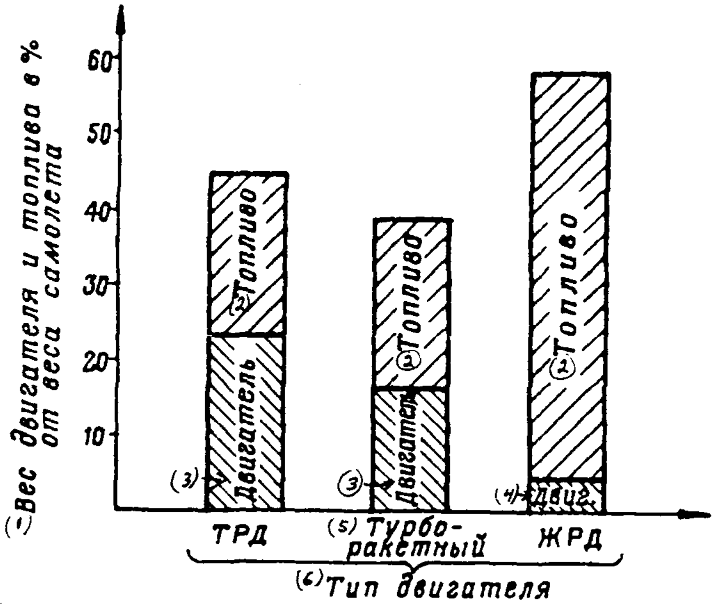

Fig. 34. Weights of engines of different types and fuel/propellant in total weight of aircraft, requiring for implementation programs of the flight given in Fig. 33.

Key: (1). Weight of engine and fuel/propellant in % of the weight of the aircraft. (2). Fuel/propellant. (3). Engine. (4). Eng. (5). Turborocket. (6). Type of engine.

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This means that the temperature of air in the compressor and after the last compressor stages in it is raised less than that in the TRD. This gives grounds to assume that a turborocket engine can be utilized at higher speeds than the TRD, prepared from the same materials. Difference in the temperature conditions will make it possible to increase the maximum speed of flight on the engine, which has parts from titanium, from M=2.5 to M=3.5, and from nickel alloys - from

M=4.5 to M=4.8.

The necessary accessories of the hypersonic direct-flow turbine, turborocket and other compound engines, which use air as working medium/propellant, are variable inlet and jet nozzle. Operating temperatures in the turborocket engines are very great. Therefore, for displacing these elements there are utilized pneumatic engines and worm mechanisms, and not electrical or hydraulic drives.

Hypersonic engines, which use air as working medium/propellant, can work only on high-energy propellants. Great prospects for hypersonic engines open up the application as the fuel/propellant of liquid hydrogen. It has approximately 2.7 times calorific value, good characteristics of burning greater than kerosene. Its density is relatively small - only 0.07 kg/m³, i.e., 11 times it is less than kerosene density. Consequently, during the application of liquid hydrogen the problem of its arrangement/position on the flight vehicle appears.

NUCLEAR ROCKET ENGINES.

Working rocket-motor characteristics on chemical fuels/propellants are limited by energy properties of fuels/propellants. As a result of use in the rocket engine of nuclear energy this limitation almost drops off. Here there are possibilities of using the working medium/propellants with minimum molecular weight,

and it means, with the high specific thrust. Furthermore, is eliminated the need for chemical combustion. The totality of these advantages makes nuclear rocket engines in contemporary cosmonautics with the main rival of chemical-propellant engines.

25.

..e foreign specialists assume that rocket with nuclear engine pear more rapidly than aircraft with this engine[1].

E[1]. The interest in the nuclear-propelled aircraft again is ed most recently in connection with the gain in weight of rt aircraft to 300-400 T and achievements on the protection clear radiations abroad. Therefore the given consideration is ned. ENDFOOTNOTE.

easons are reduced to the following.

Because of the need for the protection of the aircrew from n, good shading of the flight deck is required. But the weight raft considerably increases due to the weight of shadowing.

is shortcoming completely is removed in pilotless rockets and rably decreases in manned spacecraft because of shadow ng, i.e., to protection of flight deck only on one side - from arrangement of nuclear reactor.

2. For aircraft it is necessary to create such constructions/designs, which would have prolonged service life at high temperatures and nuclear radiation. This problem in the case of the creation of rockets is simplified as a result of the short service life of nuclear rocket engine.

3. Difficulty of ground handling of aircraft with nuclear engine. The rocket with the nuclear engine is utilized only once in view of the fact that its ground handling is considerably simplified in comparison with the servicing of autonomous aircraft.

Nuclear rocket engines, as the foreign press notes, can be used for delivery/procurement of different loads to Moon, for starting/launching of automatic interplanetary stations and for interplanetary flights of space vehicles with people. The development of nuclear reactors is initiated more than ten years ago. At present are already created and tested from the first samples.

Work on study of nuclear reactors for space vehicles occurs in several directions. The YaRD with the solid-propellant reactor, with the reactors on the dustlike, liquid and gaseous fuels are studied. The design of the YaRD with solid fuel is most fully investigated. This design is given in Fig. 35.

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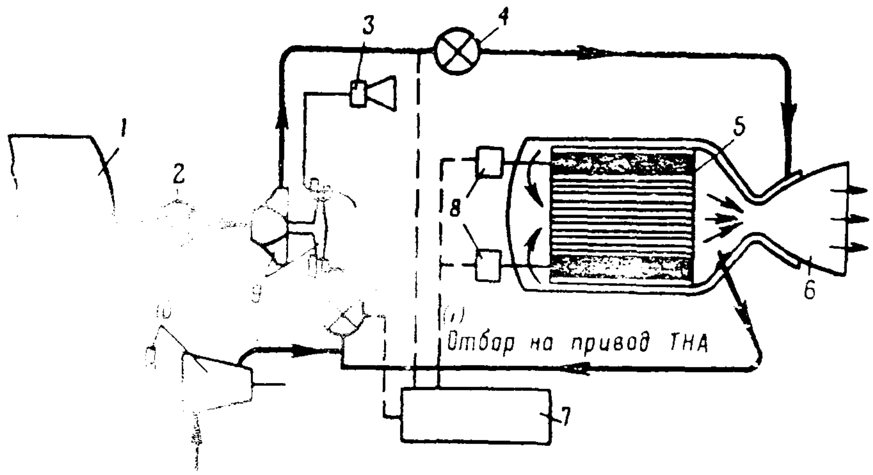
The YaRD with solid fuel consists of a reactor 5, turbopump unit 9, nozzle 6, tank 1 with working medium/propellant, equipment of control and other units. Reactor core is comprised of the solid fuel elements. Engine is equipped with the control system and of rea[illegible] scram system, with the feed sys[illegible] [illegible] some auxiliary elements.

The working medium in this case is hydrogen, and it is supplied from tank 1 with turbopump unit 9 into cooling system of the nozzle and reactor vessel, and then into reactor 5. In the reactor core working medium is heated and, being exp[illegible] in the nozzle, it e[illegible] from it in the atmosphere. In this case a reactive thrust is created. The temperature of the working medium at the output from the reactor is limited in this diagram to the heat resistance of fuel elements (2730-2930°C). Specific thrust does not exceed 800-1000 s.

According to this diagram in the USA according to the program "Rover" there is created a nuclear rocket engine "Nerva". The development of the engine was initiated in 1965. Its reactor is th[illegible] cylindrical housing, in which there are [illegible] lamellar uranium-graphite fuel elements [illegible]



[illegible] Diagram [illegible] solid-propellant YaRD: 1 - tank with working medium/propellant - hydrogen; 2 - check valve; 3 - engine nozzle of correction; 4 - control valves; 5 - reactor; 6 - nozzle [illegible] control; 8 - drives of controlling drums; 9 - turbo[illegible] (TNA); 10 - starter unit.

Key: (1). Take-off of drive of TNA.

In the center of the [illegible] region [illegible] the rods of [illegible] protection system are placed, they are cool[illegible] The power of reactor is approximately 70 M[illegible]

[illegible]sequently, into 1965 and 1966 were constructed m[illegible] [illegible] of this type with a number of design changes [illegible] [illegible] MW. [illegible] provided the possibility of full [illegible] operation during 10 min. The general view of the Y[illegible]

given in Fig. 36.

In the USA there are now developed also solid-propellant reac
of type "Phoebus". In the reactor "Phoebus-2" they intend to obta
the power of 5000 MW, the consumption of hydrogen is 130 kgf/s, th
temperature of hydrogen at the output from the reactor is 2230°C.
These parameters of the reactor will make possible to create a YaF
with a thrust of 100-120 T and a specific thrust of about 800 s.

The creation of materials, which maintain high a temperature
active region/core, is a serious problem during the development of
reactors. The efforts/forces of scientists and designers are now
directed toward the search for such high-temperature (strength)
materials, which would work at temperatures of 2730-2930°C.

Diagrams of YaRD on dustlike [powdered] and liquid propellant
principle are similar to the design of YaRD on solid fuel. Diffe
consists only of the device/equipment of reactors themselves.

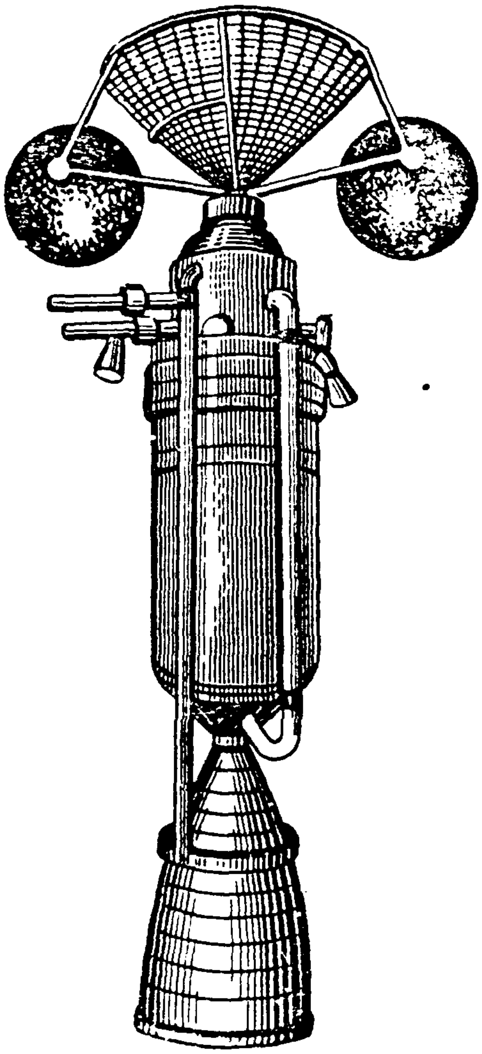


Fig. 36. Nuclear jet engine "Nerva".

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In foreign press it is noted that during use of YaRD on powdered fuel/propellant it is possible to obtain specific thrust 1000-1100 s and more, and on liquid propellant - 1200 s and more.

It should be noted that reactors with pulverulent and liquid

nuclear fuel more complicated by construction/design than solid-propellant. Therefore they are scientific they discuss about that, it is worthwhile to develop/process them. Not to concentrate all efforts during the development of solid-propellant YaRD, especially as the increment in the specific thrust during their application is comparatively small.

Specialists give considerable attention to reactors which operate on a gaseous fuel. In this diagream of the reactor it is possible not to allow the direct contact of nuclear fuel with the walls of housing and other parts of reactor. Some foreign specialists assume that in the given reactors it is possible to heat the working body to temperature of 7730-9730°C which will make possible to obtain specific thrust of 2500-3000 s and more[1].

FOOTNOTE[1]. The readers who want in more detail to be introduced to the nuclear rocket engines, the author advises to read a good article on this question of D. Zhemchuzhin and V. Ivanov, published in the journal "Aviation and Cosmonautics", No. 9, 1967. ENDFOOTNOTE.

ION-PLASMA JET ENGINES.

Recently scientists and designers intensely work at use in cosmonautics of so-called ion-plasma jet engines. In the foreign literature it is indicated that these engines will be most widespread in space flight ships.

The term "ion-plasma jet engines" for a long time was used for the designation only of such engines in which the electrically charged particles, or an electro-conductive working body, were accelerated by electrical or magnetic forces. It was considered that the kinetic energy of the rejected particles depends on the electrostatic forces (in contrast to the thermodynamic forces, which act in the rocket engines, which work on the chemical fuel/propellant).

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In recent years of experiment they were concentrated at other processes, in which the electrical energy is converted into the kinetic energy of outward flow. As a result of experiments with the electrical arc discharges of high intensity it is established that under the favorable conditions the heated working gas emerges from arc chamber/camera with supersonic speeds.

Ion-plasma jet engines require a separate energy source. In the rocket chemical- propellant engines thermal energy is obtained from the chemical energy of connection/communication, which is freed/released in the presence of the reaction between the fuel and the oxidizer. The task of designer is limited here to the selection of propellant components with the high reaction energy, for example kerosene - oxygen or hydrogen - oxygen. In the ion-plasma jet engine the separate energy source and the corresponding converting installation for the communication/report of the energy necessary

kinetically to particles of working medium/propellant must be provided for. The energy source and accelerator (converter) must be included in the engine, which develops thrust (Fig. 37).

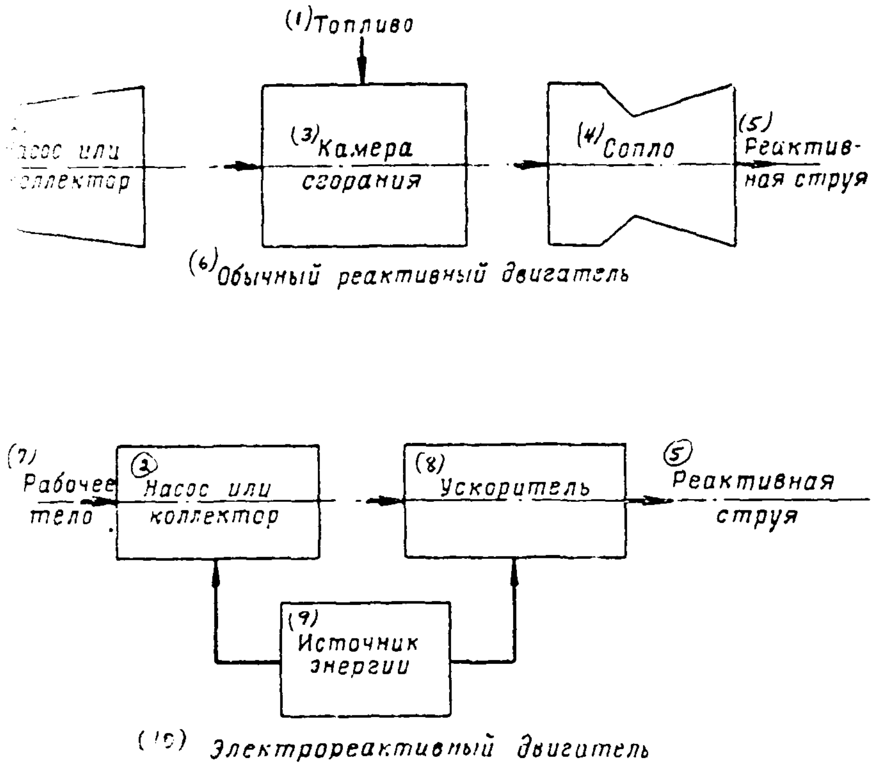

Fig. 37. Diagram of a standard jet engine and ion-plasma jet engine. Key: (1). Fuel/propellant. (2). Pump or collector. (3). Combustion chamber. (4). Nozzle. (5). Exhaust jet. (6). Standard jet engine. (7). Working body. (8). Accelerator. (9). Energy source. (10). Ion-plasma jet engine.

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Exhaust gas velocity in rocket chemical-propellant engine is limited by calorific value of reaction between fuel and oxidizer. The fuels/propellants which possess the greatest reserve of energy, for example, hydrogen - oxygen or hydrogen - fluorine, make it possible to obtain the discharge velocity on the order of 4-5 km/s. Engine with nuclear reactor can ensure 2 times, electrothermal engines - 4-5

times, and ionic and plasma - 10-100 times greater discharge velocity.

The need for production and energy conversion in separate devices/equipment limits power, which is communicated to working medium/propellant. Therefore the mass flow rate of working medium/propellant must be supported at a comparatively low level. This limitation leads to the small values of thrust and accelerations, characteristic for the ion-plasma jet engines. However, also accelerations can be supported in the long period, so that flight vehicles with these engines can reach high final speed - to 200 km/s.

Small accelerations, created by ion-plasma jet engines, are made with their with those not applied, if flight is accomplished/realized in space, where there is noticeable aerodynamic drag of apparatus, or if for changing trajectory it is necessary to overcome gravitational forces. But it is possible to utilize them successfully in space where there is no resistance to motion.

Many elements of this engine, for example conduits/manifolds, tanks, shields and radiator, apparently, will serve also as load-bearing elements of ship. Certainly, small unmanned space vehicle will differ significantly from the manned ship, intended for the flight to Mars, but in both cases engine, load-bearing elements and payload are lobate to compose the single interconnected complex.

Fig. 38 schematically shows spacecraft with ion-plasma jet

engine. Nuclear reactor is utilized as the energy source on it.

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Into payload of ship enter the system of control and navigation, communication equipment, small auxiliary source of solar energy, scientific instruments, life support system, equipment for launch and landing. Crew members are also considered as the payload.

Into device/equipment for obtaining thrust enter reactor, heat exchanger, turbine, generator, accelerating chamber (propeller/motor).

Shield is established/installed for protection of the crew from emission of the nuclear reactor.

Ion-plasma jet engines can be several types. Let us examine some of them.

Plasma ion-plasma jet engine. Plasma ion-plasma jet engine is the combined electrothermal engine system.

In it the thrust is created due to the expansion of hot plasma. Electric [illegible] is the energy source for the work of the engine. The [illegible] of its device/equipment is shown in Fig. [illegible].

Engine consists of cylindrical chamber/camera, at each end/lead of which are electrodes. As a negative electrode (cathode) the nozzle is utilized (nozzle). Special material of rod-shaped form is a positive electrode (anode). During the launching and operation of the

engine on electrodes direct current is supplied. With a certain potential difference the electric arc is formed.

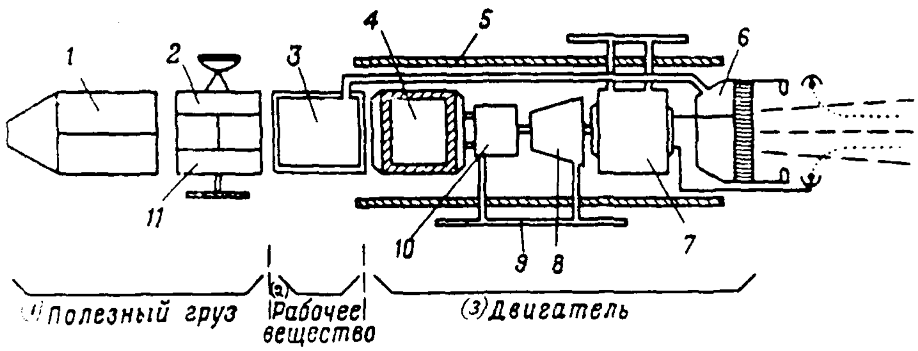

Fig. 38. Schematic of spacecraft with ion-plasma jet engine: 1 - load; 2 - system of control and navigation; 3 - work substance; 4 - reactor; 5 - load-bearing structural elements; 6 - accelerating chamber; 7 - generator; 8 - turbine; 9 - shield; 10 - heat exchanger; 11 - communication equipment.

Key: (1). Payload. (2). Work substance. (3). Engine.

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In the arc the electrical energy is expended for the ionization of atoms and the dissociation of molecules, and also to an increase in the temperature in the near-arc space. For the completion/replenishment of material, expended in the arc, positive electrode automatically is supplied. The ionized atoms and the dissociated molecules form plasma. Escaping behind the nozzle, the plasma creates a thrust force.

For the creation of a plasma engine, which operates according to this diagram, it is necessary to have coolant and to ensure protection of elements of construction/design.

n disadvantage in this engine for space flights is need for re of relatively larger quantity of working medium/propellant . Furthermore, hot plasma, acting on nozzle (nozzle), leads pid erosion of those elements of construction/design, with is contacted, if we do not take shielding measures.

This plasma engine in operating principle is very close to thermal. Therefore its specific impulse, although it is higher than in the standard ZhRD, is nevertheless limited by relatively small values.

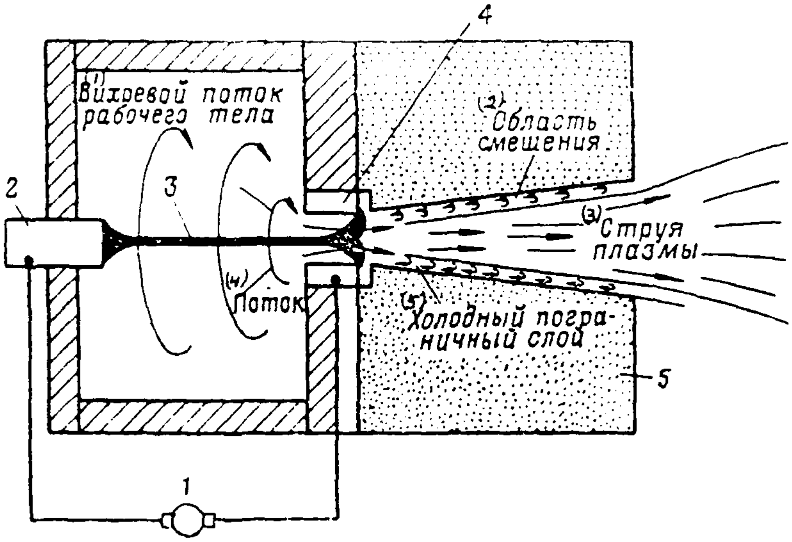

Fig. 39. Diagram of the equipment and work of plasma ion- plasma jet engine: 1 - generator of direct current of ~200 V; 2 - anode; 3 - electric arc; 4 - cathode; 5 - nozzle unit.

Key: (1). Vortex flow of working medium/propellant. (2). Mixing zone. (3). Plasma jet. (4). Flow. (5). Cold boundary layers.

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Obtaining electrical energy in plasma engines is connected with need for having bulky equipment for converting primary energy into electrical, which leads to large weights of entire power plant.

Ion engine. This engine is called still electrostatic. In it the particles of the working medium/propellant, which possess electric charge, undergo the action of accelerating forces in the electrostatic fields. The escaping particles can be the ions, charged/loaded by particles or even specks and drops. The discharge velocity of the

particles, which left the accelerating chamber, is determined by voltage drop across the ends of the chamber, [illegible] and by their mass. The transformation of electrical energy [illegible] kinetic energy of the emitted beam of particles sufficiently simply explained in the theory of electrostatic motors. However, the designers of ion engines during the creation of these engines encounter very complicated problems.

For retaining/maintaining electrical neutrality of the body of the ship the ion engine must provide outflow of strictly equal quantities of positively and negatively charged particles. The most accurate path for satisfaction of this condition - simultaneous issue of positive ions and electrons, which are freed/released in the process of ionization.

Fig. 40 gives schematic diagram of ion engine.

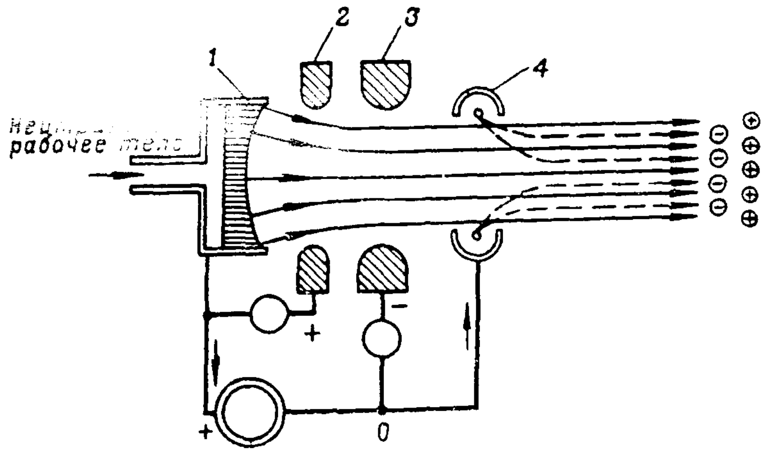

Fig. 40. Schematic diagram of ion engine: 1 - ionizer; 2 - forming electrode; 3 - accelerating electrode; 4 - neutralizer.

Key: (1). Neutral working medium.

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Its fundamental elements are storage tank of working medium/propellant, the feed system of the working medium/propellant, heater, ionizer, accelerating chamber, electron emitter and neutralizer. The supply of working medium/propellant from the tank is produced by its displacement. In the simplest case the tank is the metallic bellows, where there is retained working body in the liquid phase and whence it is displaced in the quantity, required for the feed/supply of engine. As the working medium/propellant can be used cesium, mercury and other elements. Liquid working body is passed then into the heater, where it evaporates. In the vapor phase working the body enters ionizer 1. In ionizer 1 are formed the positively charged ions[1] and negatively charged electrons.

FOOTNOTE[1]. Ion - a charged atom (or the totality of atoms); differs from normal neutral atom in terms of excess or shortcoming in one or several electrons. ENDFOOTNOTE.

These particles enter the accelerating chamber. In it under the action of the electric current, which is passed through electrodes 3, is formed electrostatic field. The forces which act in the electrostatic field accelerate the particles to very high discharge velocities - 100 km/s and more. Then the ions and electrons under the action of the neutralizer again are mixed, forming a neutral plasma.

Process of obtaining ions was studied even in the past century. In the last decade are developed the methods of obtaining the ions with the aid of the electric arc, and also the electron collision. Electric arc is considered a most effective ion source. For obtaining ions by the electric arc method the discharge between the cathode and the anode in the medium with the low pressure is utilized. The diagram of electric arc ion source is shown in Fig. 41. In this hauling gear of positive ion beam from the region of the dense plasma of electric arc it is necessary to very thoroughly shape all electrodes and magnetic field near outlet.

Ion-plasma jet engine provides greatest possibility for accomplishment of different maneuvers for midcourse to spacecraft. Aboard the spacecraft with this engine it is possible to

accomplish/realize an active guidance for the elongation/extent of entire flight from striation to the target.

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With the realization of the navigation of spacecraft with the ion-plasma jet engine it is necessary that the program of control would provide a change in the thrust vector in accordance with predetermined program of accelerations, and the coordinates of position of a ship and velocity vector continuously were compared with the values of these parameters, determined by the predetermined trajectory. The difference between the actual and given values of coordinates creates corrective commands.

As he indicates in foreign press, spacecraft with ion-plasma jet engines will have two parallel engine installations, arranged/located on certain distance one from another in transverse direction. They can diverge independently. During the cruise the thrust vectors of both installations will be parallel to the prescribed/assigned G-vector. With a maneuver accomplishment along the pitch, yaw and bank the thrust vectors will have directions, which ensure the required change in the position of ship.

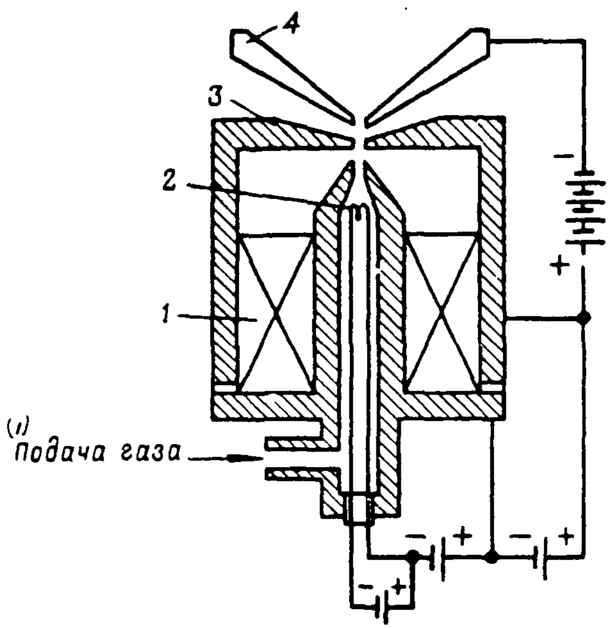

Fig. 41. Diagram of electric arc ion source: 1 - magnet coil; 2 - cathode; 3 - anode; 4 - extraction electrode.

Key: (1). Gas supply.

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The use of ion-plasma jet engines aboard manned ships, intended for studying planets, is most tempting.

Foreign specialists indicate that, apparently, an expedition to Mars can prove to be the first, aboard ships of which there will be used an ion-plasma jet engine for flight into both sides. In the press/printing is led the schematic of space vehicle with this engine for the flight to Mars, as it is presented by the foreign scientist (Fig. 42). It is assumed that this expedition will consist of five ships. In each ship three people will be located. Ships will

approach Mars and will leave in its orbit at the height/altitude of 300 km. The atmospheric drag of Mars ships they will overcome with the work of ion engines under the conditions of the reduced thrust. From the orbit the first ship will supply to Mars the means of displacement/movement on the surfaces of planet, shielded living quarters, instruments and the reserves of the necessary things; the second - small research crew.

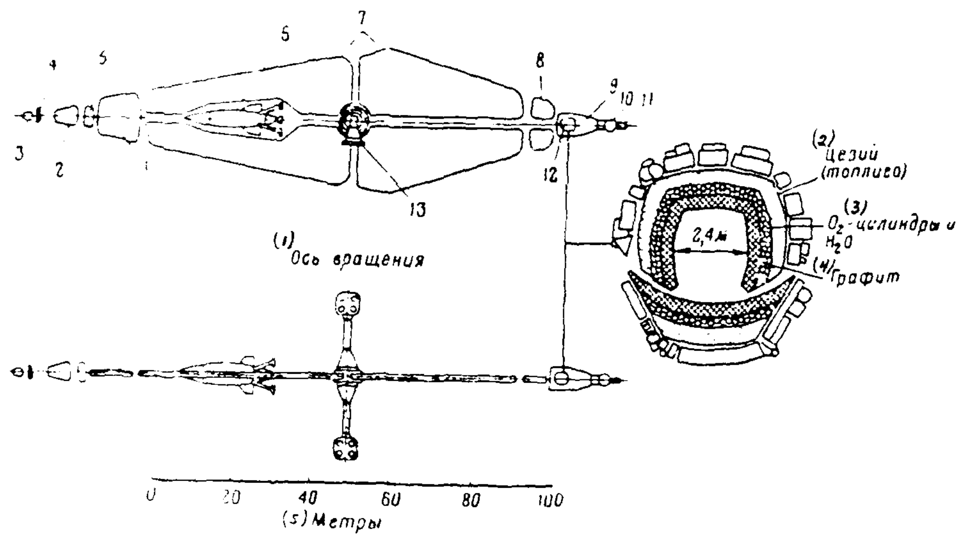

Fig. 42. Manned spacecraft with ion-plasma jet engine for flight to Mars and back: 1, 8 - cooling system of equipment; 2 - converter; 3 - reactor; 4 - shield; 5 - tank with cesium; 6 - rocket for landing on Mars; 7 - cooling system of energy converter; 9 - crew compartment; 10 - system of navigation; 11 - system, which tracks after stars; 12 - shielded cabin/compartment; 13 - ion engines.

Key: (1). Rotational axis. (2). Cesium (fuel/propellant). (3). cylinders and. (4). Graphite. (5). Meters.

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The crew of the third spacecraft is intended for rendering aid to the spacecraft crew of the second. Two ships - the fourth and the fifth - will supply to Mars fuel/propellant. Each ship will be capable is capable to return all crew members in orbit of the Earth.

By construction/design of ship will be provided for its rotation

relative to center of gravity, with which artificial gravitational force in the crew compartment will be approximately 0.1 g. It is noted that the cabin/compartment of the ship must be shielded from the penetration into it of the high energy particles of the bursts of solar radiation and emission of Earth radiation belt. The walls of cabin/compartment form thick layers of the shielding material.

It is assumed that all five ships will return to earth's orbit. In the case of failure two ships can accept onboard 15 people of crew. In the emergency case one ship can supply all participants in the expedition in orbit of the Earth.

Flight time of ships to Mars and to Earth will vice versa depend on that value of acceleration in process of flight, which they can ensure ion-plasma jet engines, and also from alignment of Earth and Mars. Flight time to Mars and will be vice versa minimum, if the flight trajectory from the Earth will be directed directly toward Mars, and the trajectory of return to the Earth will pass around the sun and will not exceed the limits of the orbit of the Earth. Taking into account latter/last condition and that that the specific installed power will be equal to 0.5 kW/kgf, the flight time of ships round trip, which includes several weeks of stay in orbit of Mars, will be approximately 560 days. The time of return flight to the Earth will be approximately two times more than flight time to Mars, since the trajectory of this flight cannot be directed directly toward the Earth: their rendezvous with the Earth without the flight around

around the sun is impossible after several weeks of the stay of ships in orbit of Mars.

RECOMBINATION RAM JETS.

There is a possibility of using in the engines an external energy source. Such a source consists of free radicals - atomic oxygen and nitrogen, that are located in the ionosphere, which begins at the height/altitude of 80 km. But how to extract their required quantity from the atmosphere?

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Indeed tne energy density in the ionosphere is extremely low. Apparently, the system intended for the catching of this energy must have a large air intake for capturing a sufficient quantity of dissociated products from the environment. The engine installation, which works due to the recombination of the dissociated gases, can be visualized in the form of the flight vehicle, which has the form of large funnel/hopper. Flight vehicle moves around the Earth at this height/altitude and with this speed, with which is possible the accumulation of the quantity of energy, sufficient for the compensation for losses to the drag and ensuring a certain maneuverability. Such engines are conventionally designated as recombination ram jets (Fig. 43). Even if this PVRD will be created, it will not nevertheless ensure the takeoff and landing of space vehicle. Aircraft launching with this engine must be produced with

the aid of the separate engine installation, similar those, which are utilized during the starting/launching artificial of the satellites, which do not have their own engine.

Thus, the recombination ram jet can only support flight, and, possibly, provide a certain maneuverability only at boundary of earth's atmosphere.

Meteoritic danger is a serious problem for space power plants.

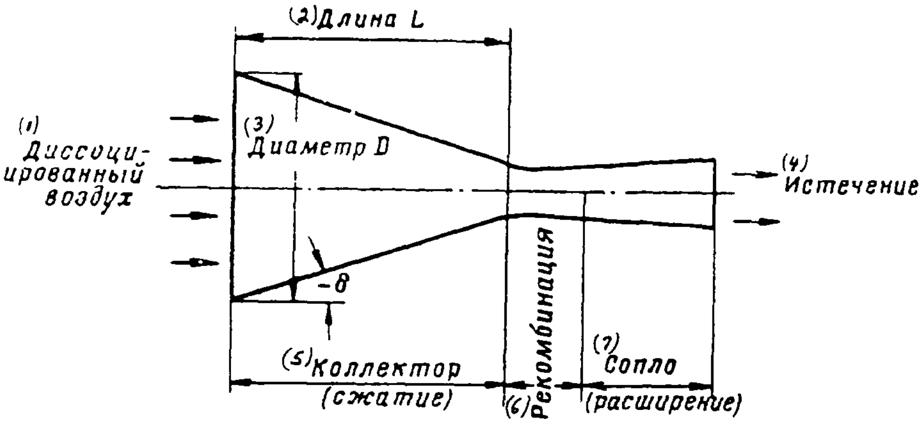

Fig. 43. Diagram of recombination ram jet.

Key: (1). Dissociated air. (2). Length L. (3). Diameter D. (4). Outflow. (5). Collector (compression). (6). Recombination. (7). Nozzle (expansion).

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What is the piercing capability of meteorites, from which materials are made the construction/design of the ship and power plant in order to avoid their damage? The study of these questions is very important. Unfortunately, the speed of the majority of the meteorites higher than speed, which it is possible to achieve and measure under laboratory conditions[1].

FOOTNOTE[1]. The average impact velocity of a meteorite against the spacecraft can be 15-30 km/s. ENDFOOTNOTE.

In press/printing difficulty of resolution of this problem of radiator-coolers of nuclear reactors, which have large sizes/dimensions, separately is noted.

ONBOARD POWER PLANTS.

As has already been indicated, the energy was necessary not only for orbital injection of the apparatus, but also for guaranteeing vital conditions to the crew, for feed/supply of equipment of means of connection/communication, etc. Therefore, the problem of energy supply in space, especially for the prolonged interplanetary flights, has exceptionally/exclusively important value.

Task of energy supply in space is most close to task of obtaining auxiliary energy on board aircraft. In both cases the weight is the determining parameter. However, in space excess weight acquires incommensurably large significance. Each excess kilogram of weight impedes output of flight vehicle into space and its subsequent maneuvers in it. Successful application of any type of space power plant depends on the ability of the energy source to develop high power with the minimum dead weight in the long period.

In space technology it is accepted to subdivide onboard energy sources into sources with reserve of energy on board apparatus and devices/equipment, which make it possible to derive energy from environment.

Sources on chemical fuel/propellant at present are one of fundamental onboard energy sources aboard spacecraft.

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Standard energy converters, used in combination with chemical

energy sources, are, as is known, piston engines, gas turbines, rocket nozzles, fuel cells and batteries comprised of them. These power plants receive the heat output of source and are converted it into the shaft horsepower of engine, either into the power of jet or into the electrical power.

But since for guaranteeing long running of similar sources (more than 24 hrs) there is required an extremely large quantity of chemical fuel/propellant, then it is not possible to utilize them virtually for many forms of space flights. However, still for a long time the chemical energy sources will be successfully used for the flights of spacecraft, especially those, which will make a landing on the surface of planets, since in this case the short duration of the work of engine during the landing makes it possible to utilize a fuel/propellant with the relatively low energy effectiveness for braking of apparatus.

In foreign press it is noted that as powerful/thick source of onboard energy will be utilized nuclear reactors, especially during endurance flights. As noted above, scientists designers develop/process nuclear rocket engines. However, in this case many problems appear. The protection of the crew of spacecraft from intense radiant fluxes and fast neutrons is one of such problems. For the protection of man it is proposed to use powerful shields. E[illegible]on has an ill effect also on some materials of the construction/design of spacecraft. Also it is necessary to shield

them. But, it appears, these and other problems in the course of time will be solved.

Sources of electrical energy. As the electric power sources can be used chemical batteries, solar batteries, usual thermodynamic engines (piston engines, gas and steam turbines), nuclear reactors, etc.

Chemical batteries found wide use on satellites and space vehicles as primary sources of electrical energy and as its storage batteries/accumulators. Their advantage consists in the fact that they produce direct current with a comparatively small heat liberation.

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However, batteries accumulate so little electrical energy, which to utilize them for the electrical jet engines, which have the large operating time, is virtually impossible.

Solar batteries successfully were used in numerous space flights for feed/supply of equipment of artificial Earth satellites. Is converted solar energy into the electrical with the aid of the thermoelectric and photoelectric processes. The concentration of solar energy is produced with the aid of the mirrors and the optical lenses. At present, apparently, it is possible to confirm that the solar batteries are most applicable at the electrical powers to 10 kW.

For obtaining greater powers there are required the installations of the enormous sizes/dimensions, which hardly can be used aboard the spacecraft.

However, as far as the use of solar energy for ion-plasma jet engines is concerned, majority of specialists consider possibility of competition of solar energy sources with nuclear sources scarcely probable. It is noted that for guaranteeing the necessary degree of concentration of energy the almost insurmountable problems, connected with the assembly of mirrors and lenses in outer space, will arise.

The nuclear reactor is most effective source of primary energy for purpose of obtaining electrical energy for ion-plasma jet engines.

But above it was noted, that nuclear reactors have number of shortcomings. Fundamental of them - high radioactivity. Another shortcoming lies in the fact that their primary energy is thermal energy, whose conversion into the electrical is insufficiently effective[1].

FOOTNOTE [1]. In the existing installations usually not more than 30-35% thermal energy is converted into electrical energy. ENDFOOTNOTE.

This leads to the need for using large radiator-coolers for emitting remaining part of thermal energy (to 65-70%) into outer

space. Therefore of great interest is the development of such power plants, in which the primary nuclear energy would be converted into electrical directly, without the use of thermodynamic processes.

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One should, however, say that the application of nuclear reactors as the source of primary energy for its subsequent conversion into the electrical - matter, apparently, of a comparatively distant prospect.

In the next two or three decades aboard the spacecraft as a source of the primary energy will be as before utilized usual thermodynamic engines (piston engines, gas and steam turbines), which will actuate turbogenerators, which produce electric power.

Very promising energy sources for space vehicles are fuel cells, in which chemical energy of fuel/propellant (for example, oxygen and of hydrogen) is converted into electrical.

Different sources of electrical energy in space vehicles are given in Fig. 44. Here there is indicated the duration of their use in time.

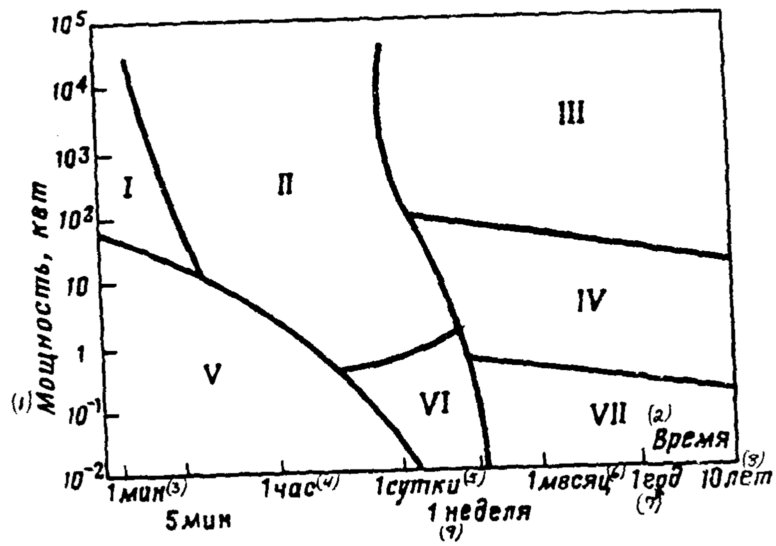

Fig. 44. Sources of electrical energy in space vehicles and duration of their use on time: I - chemical sources; II - engines which use expansion of combustion products; III - sources, which use nuclear fission; IV - sources which use nuclear fission and radioisotopes; V - battery; VI - fuel cells; VII - solar cells and radioisotopes.

Key: (1). Power, KVM. (2). Time. (3). min. (4). hour. (5). day. (6). month. (7). year. (8). years. (9). week.

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Chapter 3.

MANNED SPACECRAFT.

SOVIET SPACECRAFT.

In the Soviet Union flights of experimental satellite vehicles were begun as long ago as 1960. From May 1960 through March 1961 there were realized the experimental launchings of five space satellite vehicles without a man aboard.

Careful preliminary adjustment of satellite vehicles ensured the complete success of first manned flight into outer space. On 12 April 1961, there was put into orbit the Earth satellite "Vostok" spacecraft (Fig. 45, 46) with pilot- cosmonaut Yu. A. Gagarin aboard.

With each time the duration of flight of Soviet cosmonauts aboard ships of the type "Vostok" increased. Group flights of the ships were successfully realized.

Simultaneously with these flights in the Soviet Union work is conducted on the creation of multiplace "Voskhod" spacecraft. Improved were systems of conditioning and regeneration, and the system of soft landing, and there was provided high reliability of hermetic sealing/pressurization of the ship, which made it possible to switch

over to manned space flights without the shielding pressure suits.

Aboard ship "Voskhod-1" completed flight cosmonauts V. M. Komarov, K. P. Feoktistov, B. B. Yegorov. In the flight valuable data about outer space were obtained.

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In 1965 in orbit is derived ship "Voskhod-2", piloted by pilot-cosmonauts P. I. Belyayev and A. A. Leonov. One-and-a-half hours after the beginning of the flight cosmonaut A. A. Leonov went into open space and was in it for 12 min (Fig. 47).

"Vostok" spacecraft.

Space satellite vehicle "Vostok" is controlled apparatus (Fig. 46). Control of space ship "Vostok" can be accomplished/realized both automatically and by hand - by a cosmonaut.

Ship consists of pressurized cabin, in which is situated cosmonaut, instrument compartment, in which is placed equipment, and retro-engine installations.

The cabin/compartment is equipped with two rapidly opening hatches. On the outer side it has the special thermal insulation, which prevents the structure from the effect of high temperatures in the phase of descent.

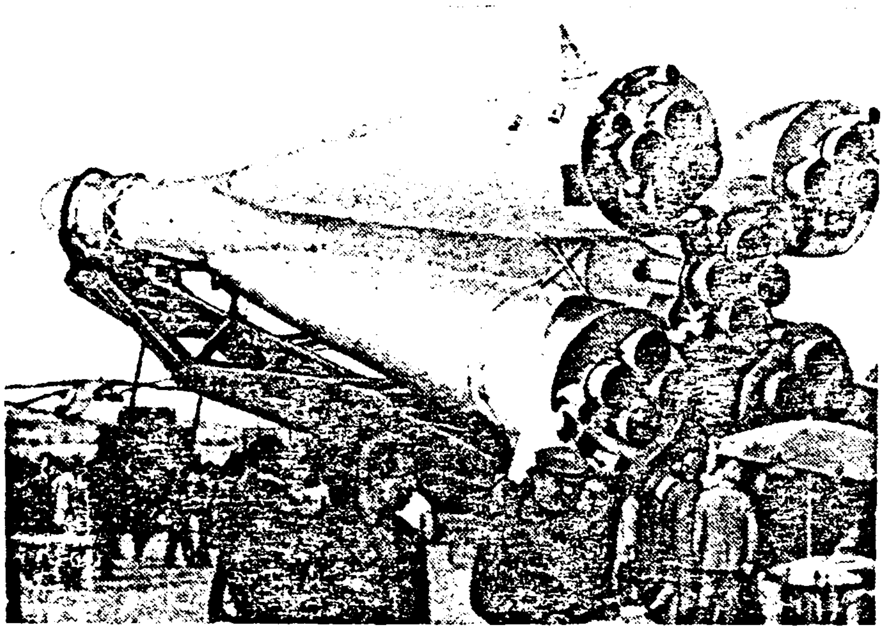

Fig. 45. Carrier rocket "Vostok" at an international aviation exhibition in Paris.

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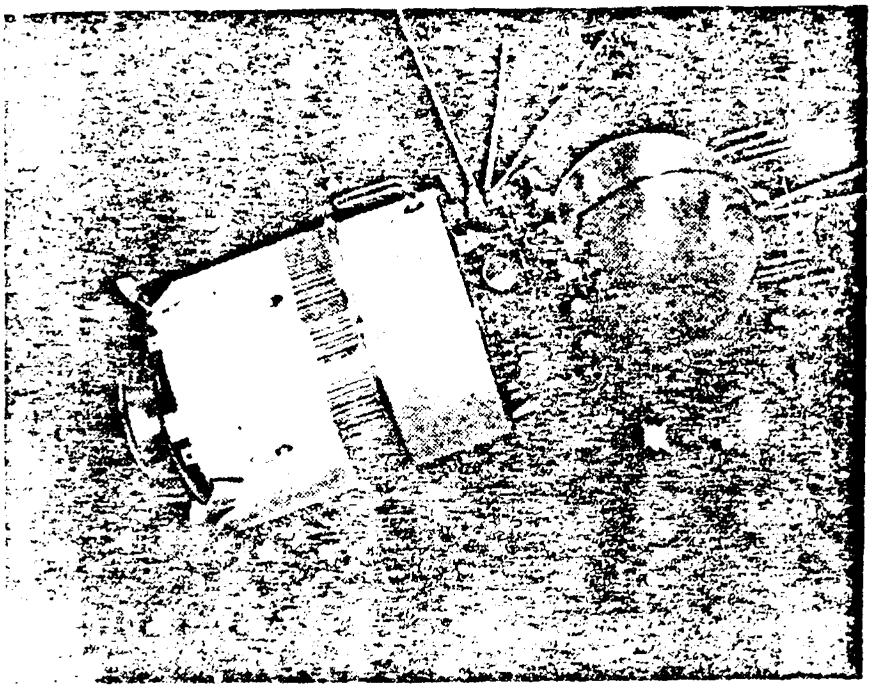

Fig. 46. Spacecraft of type "Vostok" with last booster stage.

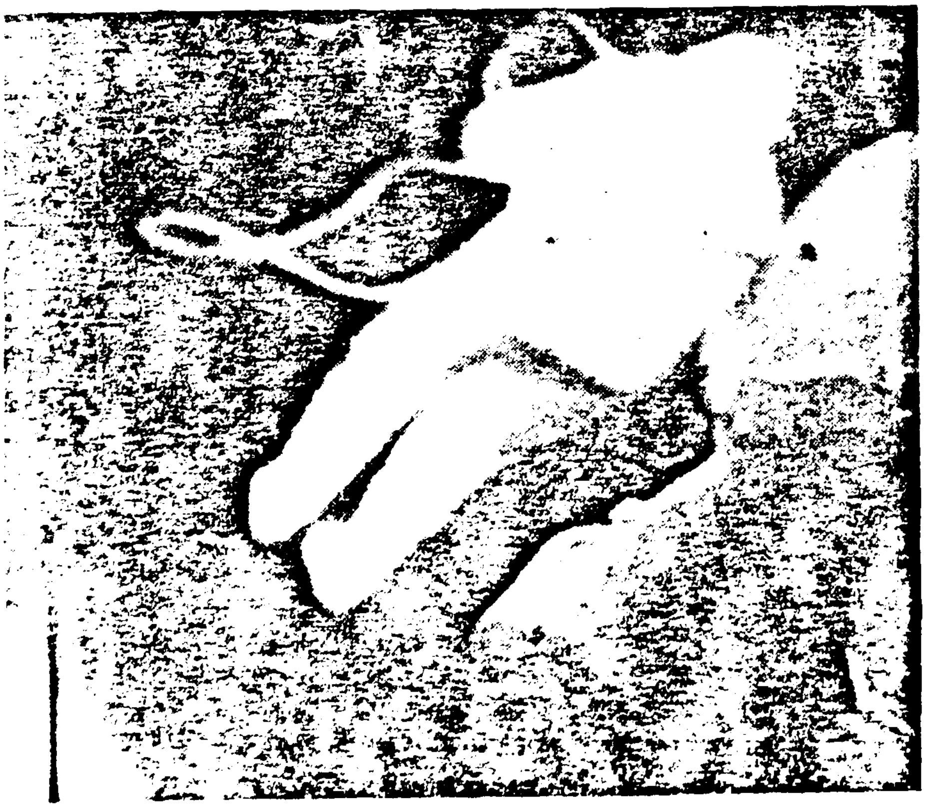

Fig 47.

[illegible]smonaut A. Leonov in outer space (photograph from [illegible]screen).

[illegible]servation in the cabin/compartment there are three [illegible]s with the high-temperature (strength) glasses, designed [illegible]ct of high temperatures, which appear on the surface of [illegible]pon the atmospheric entry. Illuminators are equipped with [illegible]ich can be opened and closed both from an electric drive [illegible]

[illegible]cabin/compartment of the ship there is placed equipment of [illegible] systems, control, supplies of food and water, part of [illegible]f radio equipment, binoculars for observation by [illegible]ough illuminators, television cameras, movie camera for [illegible]through illuminators of ship and movie camera for [illegible]within ship.

[illegible]trument compartment it is located radio equipment, [illegible]for control, temperature control system of ship, sources of [illegible]al energy.

[illegible]ring flight cosmonaut is located in catapult seat, which is his work [illegible]e and can serve as means of abandoning cabin/compartment before [illegible]touchdown. In the seat the parachute systems, which ensure the

touchdown of cosmonaut, are arranged/located.

In the case of splashdown cosmonaut can use inflatable boat, which develops automatically. If for any reasons the cosmonaut cannot use a boat, then the pressure suit will support him on the water.

During orbital flight of spacecraft pressure suit of cosmonaut is ventilated by air of cabin/compartment. In the case of loss of cabin pressure the auto-magic hermetic sealing/pressurization of pressure suit occurs. The supply of cosmonaut by oxygen and the ventilation of pressure suit in this case they are accomplished/realized by an autonomous system. The reserves of oxygen and air provide the possibility with the loss of cabin pressure to be connected with the Earth, to select landing place and to carry out descent to cosmonaut.

Orientation system, which is divided into automatic and manual, provides specific position of ship before retrorocket firing. For the determination as the cosmonaut of the direction of flight and local vertical line there serves the optical orientation device, installed on one of the illuminators of cabin/compartment.

The landing of the ship in a selected area can be produced by the cosmonaut with the aid of equipment for manual control of descent.

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Retro-engine installation is intended for changing in value and

...vector of speed of ship for purposes of its conversion ... Earth satellite to trajectory of descent. Its switching ... both automatically and by the cosmonaut with the aid of a manual ...ol.

On ... are installed radio equipment of two-way communication ... with ground- based points/items during prelaunch ...cedure, in track-out phase and in flight along orbit, and ...vers and two transmitters of the short-wave band, a ...ransmitter of ultrashort-wave range, magnetic tape r... broadcasting receiver, and a television system. The ... selects the form of radio communication by the start of one ...annel.

There are installed aboard the ship, furthermore, telemetering equipment ... system of autonomous air recycling; sensors, which ensure control ... of work of onboard equipment of ship in flight; radio ...", that works at frequencies 19.995 and 19.990 MHz; ... control/check and recording of physiological functions of ... flight - frequency of pulse, respiration, etc., which ...sible to conduct constant observation of state of cosmonaut. There are also following systems ...conditioning, cockpit pressure control, feed/supply and water provision thermal control in the cabin/compartment, illumination, touchdown.

Weight of the space satellite vehicle "Vostok" was 4725 kgf.

Spacecraft "Voskhod-2".

Manned two-place space satellite "Voskhod-2" has been developed on the base of the space satellite vehicle "Voskhod" for the egress of the cosmonaut from the ship into outer space by the of locking [sluicing].

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The ship consists of:

- a pressurized cabin, in which the crew is situated, there is placed equipment for life support, supplies of food and water, means of control and direction of work of onboard ship systems, part of instruments of radio equipment, television cameras, television monitoring system, cine- and photographic equipment, equipment for medical and scientific investigations, means of direction finding in sections of descent and touchdown;

- from the instrument compartment, in which there are placed instruments of radio equipment of ship, retro-engine installation, equipment of control system, temperature control system, electric power sources.

Aboard the ship there is installed a standby solid-propellant retrorocket, which duplicates the fundamental brake values for the egress of the cosmonaut into outer space and returns to ship.

For preservation from effect of high temperatures in the phase of the descent, the pressurized cabin on the outer side is covered with a special thermal insulation. In the cabin/compartment there are three hatches. The cabin/compartment is equipped with three illuminators, through which the crew can conduct visual observation, and movie and still photographing. Illuminators are equipped with the high-temperature (strength) glasses, designed for the effect of high temperatures, and have shutters.

Crew is located in cushioned seats, made on duct/contour of adjacent parts of body.

With boric section of ship is hermetically sealed container, in which is placed equipment and retro-engine installation. Outside in the section there are placed tanks with a reserve of compressed gas and orientation system engines of ship, tanks with compressed oxygen and air for the ventilation of pressure suits and supply of crew by oxygen with the emergency depressurization of cabin/compartment, the antenna systems of the radio systems of ship, the radiator of temperature control system. Upon the return of ship to the earth the instrument compartment is separated/liberated from the ship and burns in the dense layers of the atmosphere.

An airlock makes it possible for the cosmonaut to emerge from ship and to return back to it without a loss of cabin pressure. It is installed on the cabin/compartment of ship and is connected with it by

construction/design of spacecraft. Also it is necessary to shield

hich is closed by a pressurized cover. The hatch cover is ide the pressurized cabin; moreover, it is opened and closed ly by an electric drive with the aid of a special Control of the drive is produced from a panel. The cover ed and closed by hand if necessary.

ges the cosmonaut into outer space from the airlock through its upper part, equipped with a pressurized cover, which with the aid of the electric drive. In the airlock there two motion-picture cameras for photography of the process ry of cosmonaut into the chamber/camera and output from it, ng system, control panel and units of systems of the The installation of a motion-picture camera for ing the cosmonaut located in outer space, tanks with a air for the supercharging/pressurization of airlock and an emergency reserve of oxygen is provided for. The craft from the panel, installed in the cabin/compartment, controls t. If necessary the second cosmonaut from a panel installed ock can control the basic operations of locking.

trol of the ship can be both the automatic and manual. Equipment for manual control of flight and of descent makes it possible for crew to by hand orient ship in the space and to produce its landing in the selected area, utilizing in this case by any of the available brake motors: either fundamental (liquid), or standby

olid-propellant). During the manual control the crew utilizes an ptical orientation device for determining the direction of flight and cal vertical line or ionic plotters of the velocity vector of ship.

Automatic flight control equipment and of descent makes it ssible to control/guide according to predetermined program onboard stems and orientation system of ship in space and to help ship in escribed/assigned area with the aid of fundamental retro-engine stallation.

With the automatic release of ship from orbit axis/axle of engine stallation is advanced in sun by uniaxial orientation system, which ses electronic sensor of sun.

On board ship is established/installed equipment of two-way mmunication of crew with ground-based points/items during prelaunch rvicing procedure, in track-out phase and in flight along orbit, and so equipment of two-way communication of left cosmonaut with craft mmander.

Television system has two cameras of external observation, two ameras and monitors within the cabin/compartment.

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Aboard the ship there is also equipment of the control and recordings of physiological functions of the crew in flight;

telemetering equipment and system of autonomous recording; radio system "signal"; conditioning system; control system of pressure in cabin/compartment; power-supply system and water supply; temperature control system in cabin/compartment; lighting system; a system of landing, which provides a safe landing of the ship and crew with virtually zero vertical velocity.

Weight of space satellite vehicle "Voskhod-2" 5.68 T.

SPACECRAFT OF THE UNITED STATES.

Considerable work in the mastery of outer space is being carried out in the USA. In this country there were developed and launched into space manned ships "Mercury" and "Gemini". There is conducted extensive work on the creation of the "Apollo" spacecraft, which is intended for the landing of cosmonauts on the Moon.

Spacecraft "Mercury".

The space satellite vehicle "Mercury" was developed for studying the possibility of manned space flight, man's participation in control of ship, and also for an investigation of the effect on man of the space flight condition. The general view of ship is given in Fig. 48.

In accordance with requirements NASA, form of this ship was selected in the form of truncated cone, which converts into cylinder,

with small height/altitude, on which was established/installed one additional (upper) truncated cone (Fig. 49). Diameter of the base of ship is 1.88 m, the overall height is 2.74 m, the space of cabin/compartment is 1.4 m^3. The weight of the ship without the emergency recovery was 1.3-1.4 T.

In lower truncated cone in the pressurized cabin the cosmonaut is placed, in the cylindrical section parachutes are located, while in the upper truncated cone - antennas. To the upper cone with the aid of explosives bolts there is fastened a tripod of framework construction, which carries two RDTT. One of the engines is emergency and serves for the department/separation of ship from the carrier rocket and its lift to the safe height/altitude in the case of the onset of malfunctions during the starting/launching. Another engine is intended for the department/separation of tripod from the ship after the engine cutoff of carrier rocket.

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The ship is equipped with different equipment and systems necessary for the completion of the flight. Each assembly of equipment is duplicated/backed up, some systems (for example, control systems) are three. It was support their automatic operation with the fundamental task during the development of systems. However, is provided for manual control of systems, accomplished by a cosmonaut, and control according to commands/crews from the Earth.

Construction/design of ship is made in essence of their titanium. The bending, stretching and compressive loads are received by framework/body from the titanium stringers. Outside construction/design is covered/coated with a layer of insulation/isolation from ceramic fibers.

The cosmonaut in the cabin/compartment of the ship is placed on a seat. Fixation of the cosmonaut on the seat is produced with the aid of seat belts (shoulder, breast, knee and groin), and also a band. The les are held on the spot by means of the devices/equipment fastened/strengthened in the floor.

For maintaining vital conditions are provided for two systems: one for cabin/compartment, another for suit. Both systems supply pure oxygen under the pressure 0.35 kgf/cm². Oxygen intake at normal temperature and pressure is determined by 500 cm³ per minute, the liberation of moisture and water - 2.7 kgf in a 24 hour period, carbon dioxide - 1.2 kgf in a 24 hour period. The heat removal from the cabin/compartment is accepted at 100 kcal/h, the temperature in the cabin/compartment must be maintained in the range of 27-38°C.

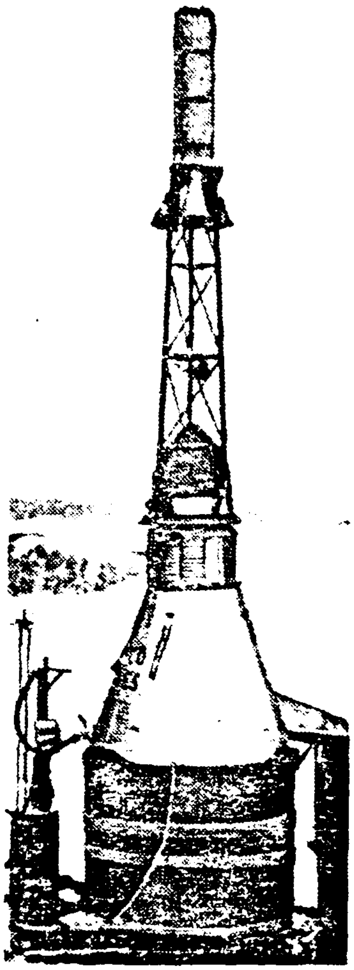

Fig. 48. General view of spacecraft "Mercury".

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Automatic stabilization and control system of position of ship is based on use of spinning gyroscopes and infra-red scanners of horizon/level. The fundamental function of system is the automatic stabilization of ship after its department/separation from the carrier rocket, and also the maintaining the necessary position of ship in the trajectory. The work of system is accomplished/realized with the aid

of 18 nozzles, which use hydrogen peroxide. Under the automatic effect of system work 12 nozzles, on four for each axis/axle. During the manual control 6 nozzles in the system are utilized.

On the satellite "Mercury" there is provided for use of two main communication systems: flight and emergency-recovery. Telemetering system during the flight can transmit 90 different data. The supply of power of all systems are six silver-zinc batteries.

Into system of touchdown of ship enter brake parachute, main parachute with circular canopy, inflatable shock-absorbing balloon and different means for the fastest detection of the landing place of the ship (the flare bomb "Safar", flashing light, radio beacon, etc.).

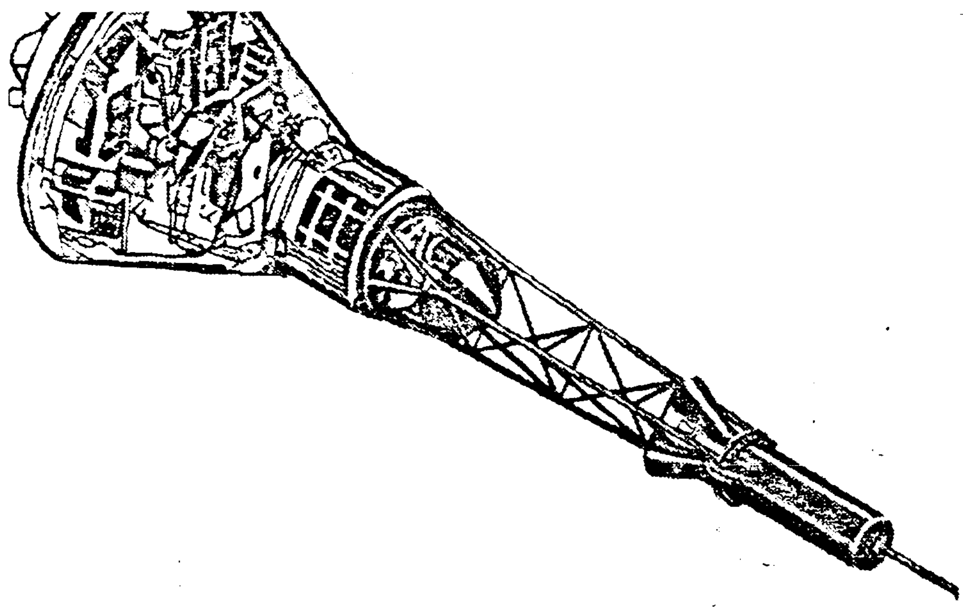

Fig. 49. Diagram of the system of spacecraft "Mercury".

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Work according to program "Mercury" was initiated in 1959 and completed in 1963. In orbit were derived four satellite vehicles "Mercury" with the cosmonauts aboard. All were placed in orbits by rockets "Atlas-D". As a result of the flights, it is established that the aged cosmonaut endures well g-forces and weightlessness, can actively participate in the control of the ship and carry out scientific observation and experiments. In the presentations of the leaders of the program "Mercury" it was indicated that as a result of works, carried out according to this program, it was possible to obtain experience in the design, construction and testing of the manned ship; for the operation of tracking stations the ship, the rescue of ship, election and to the preparation for cosmonauts,

creation and operation life-support system and biotelemetric equipment.

The general expenditures for the program "Mercury" were 380-412 million dollars.

Spacecraft "Gemini".

The space satellite vehicle "Gemini" was two-place (Fig. 50). For accomplishing the entire study program were provided for 12 its starting/launching in orbit of Earth satellite. The first ship without a crew was launched during April 1964. The launching with a crew was produced during March 1965. The ship completed three revolutions around the Earth. The flight of the ship, injected into orbit during June 1965, was most prolonged. It flew for more than 4 days.

In the USA flights of ships of this type are considered as prepatory to manned flights to Moon and creation by inhabited orbital of station.

The ship "Gemini" has an overall diameter of 3 m, height/altitude of 6.7 m, weight about 3.5 T and consists of four sections (sections). In the first section - head - are placed the radar and parachute system. The form of this section is conical. During the flights with the orbital rendezvous and the mating with the rocket "Agena", which

were accomplished/realized more lately, nose section entered into the specially established/installed tail cone of the rocket. Then the cylindrical section, where sixteen ZhRD of the orientation system, utilized for orbit ejection and entry into the atmosphere, are arranged/located goes.

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In the following section the crew is placed. The fourth section (it is called the transfer, auxiliary) serves for the connection/attachment to the carrier rocket. In it the equipment necessary for the flight and the ZhRD, utilized for orientation and maneuvering of ship in orbit is installed. This section before entry into the atmosphere is thrown off.

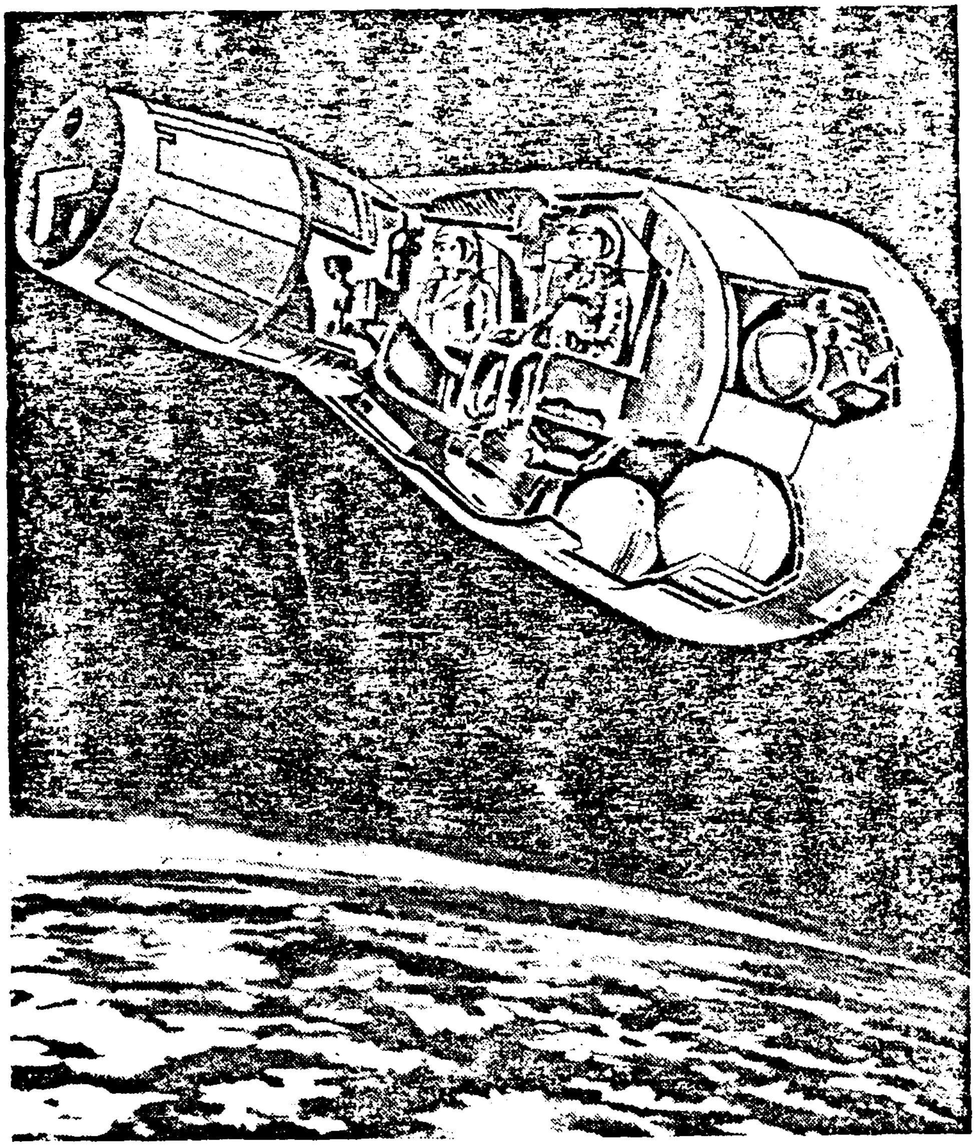

Fig. 50. Spacecraft "Gemini".

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Cockpit windows of crew is made from two separate glasses, prepared from thermoresistant material, which contains to 96% of silicon dioxide, which makes it possible to short-term maintain/withstand temperature to 1200°C.

Equipment of life-support system aboard ship "Gemini" is placed in crew compartment and in auxiliary section. The total weight of equipment for the flight, designed for 2 days, is approximately 150 kgf, and for the flight to 14 days - about 190 kgf. Nominal temperature in the cabin is 26.6°C. Cosmonauts breathe by pure oxygen. The fundamental reserve of oxygen for the respiration is stored in the liquid state in the tank, placed in the auxiliary section. For the flight by the duration of 14 days is established/installed the tank with a diameter of 50 cm, the containing 47 kgf of liquid oxygen. The supplementary reserve of oxygen is stored in the gaseous form in two tanks, which are located in the crew compartment; it is designed on 120 min, i.e., to the period, required for the completion of one turn, the atmospheric entry and landing. In the case of damaging the feed system of the fundamental reserve of oxygen the automatic changeover to the supply of oxygen from the tanks occurs.

In seat of each cosmonaut is block of two tanks/balloons with emergency reserve of oxygen, which is utilized by cosmonaut in the case of ejection/launching.

Prior to start pressure suits of cosmonauts are blown by pure oxygen.

For the decontamination of the artificial atmosphere a special

unit is used. In it are found hydroxide of lithium and filters from the activated carbon. Lithium hydroxide absorbs carbon dioxide and filters odors from the activated carbon. For the absorbtion of moisture the devices/equipment of the type of wicks serve. Purified oxygen is cooled in the heat exchanger and with the addition of fresh oxygen from the tank again is utilized for the respiration.

It should be noted that application of pure oxygen in artificial spacecraft atmosphere is dangerous in fire sense. It already led to the tragic death in 1966 of three American astronauts aboard the "Apollo" spacecraft.

Drinking water is stored in a small tank installed in crew compartment.

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Prior to the start the small tank fills upward, and then in proportion to consumption is supplemented by water, which is formed as a result of reacting of hydrogen and oxygen in the fuel cells, and by the water, condensed from the vapors, which are contained in the artificial atmosphere. In the presence of the reaction of hydrogen and oxygen there is formed up to 227 gf of water an hour, while during the condensation of vapors - to 277 gf.

The life-support system, which makes it possible for cosmonauts to emerge from ship located in orbit, must have weight according to

American data not more than 11 kgf. In the case of the disturbance/breakdown of the airtightness of the pressure suit of cosmonaut the system must during 5 min provide nominal pressure and nominal oxygen flow in the pressure suit.

With the onset of the emergency situation at the start or in initial phase of flight (to height/altitude of 21 km) cosmonauts must be catapulted in seats and go down by parachutes. If emergency situation arises at the height/altitude of more than 21 km to the stage separation of carrier rocket with the aid of the retro-engine installation (TDU), after which it will continue ballistic trajectory flight. At the upper point in the trajectory from the ship the section of the TDU, which provided its stabilization, will be separated/liberated to the given moment. In this case the entry of ship into the dense layers of the atmosphere and landing will pass just as with the normal orbit ejection.

If emergency situation appears after stage separation of carrier rocket, then to cessation/discontinuation of work of second-stage engine cosmonauts must turn off this engine and with the aid of retro-engine installation separate/liberate ship from booster stage. In this case entry into the atmosphere and landing are accomplished/realized as with the normal orbit ejection.

Ejection is produced with ie aid of explosive charges. For the ejection it is necessary to extend ring from the container installed

between the legs. Interlock system prevents the operation of explosive charges before the hatches will be opened, through which there are thrown out seats with the cosmonauts. During the operation of explosive charge the seat moves along the rails, which are for it guides. After the output of seat with the cosmonaut from the ship is included the that installed in seat of RDTT, which throws it forward. Maximum acceleration during the ejection is 24 g.

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In accordance with calculation data, the RDTT in the case of the emergency of rocket with the start it must ensure the rejection of seat with the cosmonaut to the side from the rocket on 150 m. After the rejection of seat the cosmonaut is separated/liberated from it and lowers to the earth by the parachute.

System of control and orientation of ship "Gemini" inertial (Fig. 51). It provides control upon the rendezvous of ships in orbit, the descent of ship from the orbit, entry into the atmosphere and with descent to the earth. For the flight control of carrier rocket is a radar system, but in the case of its output from the system controls of rocket automatically are connected to the inertial system for control of ship. Control of ship is produced from the panel, established/installed in the cabin/compartment. On the panel are placed the indicators of distance to missile-target and rate of closure with it, and also the indicator of orientation. The control stick of the orientation system of ship is located so that both

cosmonauts can control it.

Orientation system of ship consists of fundamental and slave units of gyroscopes, power converter and two miniature computers. The actuating elements of the orientation system and control ship are thirty-two ZhRD, which operate on the hypergolic fuel, which consists of monomethyl hydrazine (combustible) and nitrogen tetroxide (oxidizer). Helium is used for the fuel-tank pressurization. The specific thrust of these engines in connection with the application of an more effective fuel/propellant is considerably higher than the specific thrust of engines, it is earlier than used aboard the spacecraft of the type "Mercury". Sixteen ZhRD of the orientation system and maneuvering in orbit are located in the second (cylindrical) section by two units in the circumference of the housing. They provide orientation of the ship upon entry into the atmosphere and with descent to the earth. The engines of each block are established/installed in pairs. The thrust vectors of dual engines are directed tangentially toward the lateral surface of the housing of section in the opposite directions. The orientation of ship completely provide the engines of one block, the engines of another block - spare. Orientation begins approximately in 5 min prior to entry into the atmosphere.

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The ZhRD are controlled automatically on the commands produced by computer. The automatic system of engine control of both blocks it

was necessary to introduce because in the period of entry into the atmosphere the cosmonauts experience very heavy overloads and they cannot control/guide ship.

Eight onboard ZhRD of orientation system, which have thrust 11.35 kgf, are placed in auxiliary section. They are brought to four blocks. In pressure feed system into the combustion chambers of these engines compressed helium is utilized. The craft commander controls/guides all engines. Two additional ZhRD, which have thrust of 45.4 kgf, are placed in the auxiliary section. Remaining six ZhRD of this system are established/installed in the crew compartment.

The retro-engine installation consists of four RDTT, which develop thrust of 1135 kgf each.

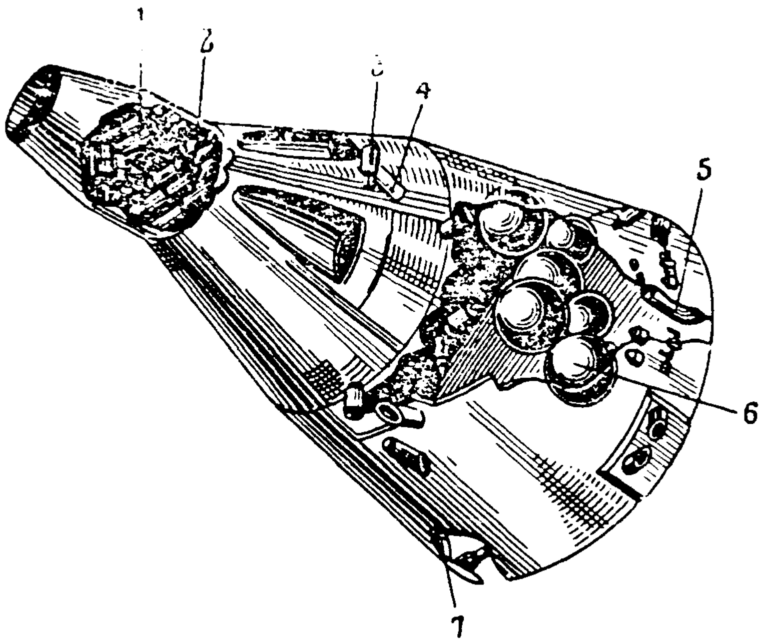

Fig. 51. Diagram of layout of elements of orientation system and maneuvering of spacecraft "Gemini": 1 - ZhRD with thrust of 11.35 kgf (only 16 engines); 2 - two fuel tanks, two oxidizer tanks and two tanks with helium (for feed of front ZhRD); 3 - ZhRD with thrust of 45.4 kgf (total of 4 engines); 4 - two ZhRD with thrust of 38.7 kgf; 5 - two ZhRD with thrust of 45.4 kgf; 6 - two fuel tanks, two oxidizer tanks and two tanks with helium (for feed of aft ZhRD), 7 - ZhRD with thrust of 11.35 kgf (total of 8 engines).

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Engines are installed on the frame from the aluminum alloy. For guaranteeing the descent of ship from the orbit the engines must be connected in series so that the next engine would begin operating earlier than the preceding/previous will be disconnected.

Simultaneously all RDTT are switched on only for the separation of ship from the carrier rocket in the emergency situation.

For guaranteeing of two-way radiotelephone circuit, transmisson of telemetry data and reception of commands, aboard the ship the corresponding radio equipment, which operates in different ranges, is installed. Flight program "Gemini" was assumed the cessation/discontinuation of radiotelephone circuit with the cosmonauts and the period of the entry of ship in the atmosphere. It was communicated also that in the telemetering ship system will be used two devices/equipment: one for the recording of information from the sensors, placed in the pressure suits of cosmonauts; the second - for the recording of the parameters of ship systems.

Radar, established/installed aboard ship, is intended for guaranteeing orbital rendezvous. Its equipment weighs about 32 kgf and is placed in the nose section of the ship. Radar has viewing angles to 70° in two planes. With its aid on the phase shift of the signal, obtained from the radar transponder on the rocket "Agena", was determined the direction, and on the time, which passed from the impulse/transmission of signal to the reception of response signal, distance to the rocket. These data, introduced into the computer, make it possible to calculate the necessary maneuver for the rendezvous. The radar began to work during the approach with the rocket, when the distance between the ship and the rocket was decreased to 460 km.

Technical assignment provided for, that mean time between two failures of radar must be not less than 1000 hour.

Ships "Gemini" were started by two-stage carrier rockets. To the output of ship for orbit the parameters of the trajectory of carrier rocket it was designed by ground-based computer. Then they were introduced into the airborne computer so that at any moment control of rocket could be transmitted to the onboard inertial system for control of ship.

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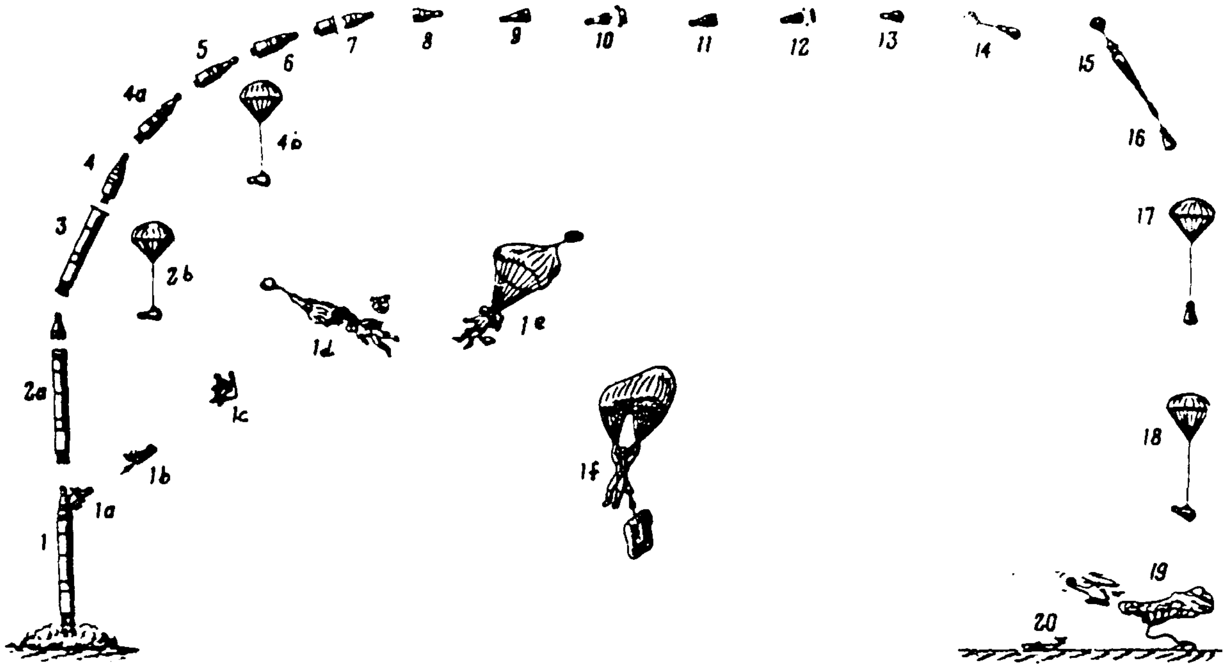

Fig. 52. Diagram of takeoff, emergency descent, entry into the atmosphere, and normal landing of the spacecraft "Gemini": 1 - launch of carrier rocket (4 s after switching on ZhRD of first stage); 1a - possible ejection of seat of cosmonaut in the case of emergency immediately after launch; 1b - termination of work of catapult rocket; 1c - separation of seat; 1d - production of brake parachute; 1e - production of main parachute; 1f - release of lifeboat and food reserve; 2a - possible separation of ship from carrier rocket in the case of emergency at altitudes of 4500-24000 m; 2b - descent of ship by parachute; 3 - separation of used-up first stage; 4 - switching on of ZhRD of second stage; 4a - possible separation of ship in the case of emergency at altitudes of more than 24000 m; 4b - descent of ship

by parachute from high altitude; 5 - termination of operation of ZhRD of the second stage; 6 - dropping of nose fairing; 7 - separation of the used-up second stage; 8 - output of ship in orbit; 9 - rotation of ship by base/root (transfer section) forward; 10 - dropping of transfer section; 11 - retrorocket firing (TDU); 12 - dropping of section of TDU; 13 - beginning of entry into atmosphere; 14 - deployment of brake parachute (altitude of 15000 m); 15 - deployment of drogue chute (3250 m); 16 - deployment of main parachute and the dropping of head section; 17 - complete opening of main parachute canopy (3000 m); 18 - conversion of ship into the suspension at two points; 19 - splashdown and the dropping of system; 20 - delivery of crew to aircraft carrier.

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In the vertical trajectory phase preset angle of flight was established/installed by rotation of carrier rocket relative to longitudinal axis. Before the switching on of the second-stage engine of command/crew the executive controls they entered on the radio channels from the Earth. The airborne computer of the inertial system for control of ship simultaneously issued commands/crews. On to the executive controls of rocket these commands/crews entered only after the switching on of second-stage engine. According to the data of flight speed, refined with the Earth, the computer of ship were designed not only cutoff point of second-stage engine, but also maneuvers, necessary for the trajectory correction with the aid of the onboard engines, if it proved to be that the ship left in orbit, which differs from calculated. Ship was separated/liberated from the second stage with the aid of the onboard engines.

Fig. 52 shows the diagram of takeoff, emergency descent, entry into the atmosphere and splashdowns of spacecraft "Gemini". Initially, besides splashdowns, which were produced in the western part of Atlantic Ocean, it was proposed to fulfill several landings on the dry land with the aid of the flexible wing or parachute system, and to also test the possibility of soft landing on the dry land with the aid of the parachute system and the brake RDTT.

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However, according to the reports to the foreign press, landings on

the dry land will be hardly accomplished/realized.

One of special features of spacecraft "Gemini" is system of power supply, in which are used oxygen-hydrogen fuel cells, designed for work under conditions of weightlessness. To them continuously are supplied oxygen and hydrogen, which are stored in the liquid state in the spherical tanks/balloons. During the heating to 60° they pass into the gaseous state and are fed/conducted to the fuel cells. An increase in the temperature is provided by heater - the gold-plated filament, wound around the internal spherical shell of tank/balloon. Fuel cells and tanks/balloons are placed in the transfer section of ship.

In foreign press it is indicated that sources of power supplies, in which are used fuel cells, five times of more easily usual silver-zinc storage batteries.

In one of flights of ship "Gemini" to cosmonauts was given assignment meet container, which generates orbital flight. Cosmonauts made a series of operations for guaranteeing the rendezvous, but the part of fuel cells worked poorly and did not provide in a sufficient measure ship with electric power. Therefore to crew it was ordered not satisfy maneuvers for the approach with the container and in every way possible economize electric power. Cosmonauts reported, that they prior to the eighth turn periodically observed container. One time container proved to be at a distance about 300 m from the ship, but

attempts at the approach with it for the reasons indicated were not undertaken.

Power supply was improved on orbit 35 and cosmonauts obtained assignment to simulate orbital rendezvous with conditional rocket, i.e., to derive spacecraft into fixed point of space, in which allegedly must be located this rocket. Cosmonauts began maneuvering and in 2 hours 15 min decreased apogee altitude of orbit on 19-20 km, they increased perigee altitude on 16 km, changed flight course, after transferring ship into the plane, which coincides with the plane of the orbit of conditional rocket, then they increased apogee altitude they put ship into the orbit, which coincides with the conditional orbit of rocket.

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During the maneuvering the data of airborne radar and computer were utilized.

As noted in foreign press, during flight cosmonauts observed starting/launching of ballistic missiles, made photographs of aircraft carrier, revealed/detected typhoon and determined its location.

In connection with some malfunctions of ship systems the solution about its landing was accepted. The landing was completed after 122 revolutions around the Earth, and splashdown occurred with an undershoot of 165 km to the prescribed/assigned zone, where the

ensuring ships were located. During the landing there was first released brake parachute, and after 6 min - the main one.

During flight in cabin/compartment of ship "Gemini" there was maintained a temperature of 18.3-21.1°C.

In one of subsequent flights was realized rendezvous of ship "Gemini" with specially launched rocket stage "Agena" and mating with it. Was at first derived rocket stage "Agena" in a circular orbit by an altitude of 298 km. After 1 hour 41 min by the carrier rocket "Titan" there was launched into an elliptic orbit the spacecraft "Gemini" with two cosmonauts. On the third orbit the flight trajectory of ship was changed to circular with an altitude of 270 km. When ship was located at a distance of 275 km from the step/stage, crew switched on radar of rendezvous, then it switched the onboard central computer (TsVM [digital computer]) to the execution of final rendezvous maneuver and began checking the ship systems, which ensure rendezvous and mating. Maneuvering with the aid of the engine, cosmonauts began to feed the ship to the rocket stage "Agena".

Final stage of rendezvous was accomplished/realized in shadow of Earth, since this facilitated observation of signal lights of the stage "Agena". Twice in this stage the spacecraft crew produced the correction of speed. During the second correction the distance between the apparatuses was reduced to 7 km and approach became possible not only with the aid of the locator, but also it is visual.

Ship passed under the rocket stage "Agena" and passed it. After this, the cosmonauts with the aid of the stability-guidance jets turned ship on 180° relative to the center of gravity.

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Then followed braking ship by engine, as a result of which of the speed of ship and stage "Agena" they were equalized. The distance between the apparatuses reached 15 m and did not change during 45 min. The subsequent braking made it possible to carry out normal mating of ship and rocket step/stage. But tangential control motors were switched on due to the malfunction of onboard electrical system and the high-spin motion of the butted apparatuses around the longitudinal axis was begun. It was necessary to divide apparatuses, for were utilized brake rocket engines, and to forego the outlined program of experiments. After the separation of the butted apparatuses it was possible to stabilize ship and to carry out a landing. Step/stage "Agena" remained in orbit.

Program of experiments provided for output of one cosmonaut of ship. Pressure in the cabin/compartment had to be lowered to 0.25 kgf/cm^2, and cabin/compartment - is completely unsealed. Then one of the cosmonauts had to leave the hatch of ship and take several photographs of the surrounding space, reach the docking module and return to the cabin/compartment for the overcharging of the chamber/camera of camera. After this, it again had to leave into open space and during 45 min make a number of operations with the aid of

the instruments. Only after this it was proposed to divide to apparatus. Afterward rocket "Agena" had to be located at a distance about 20 m from the ship. In this case the cosmonaut could reach it utilizing for the movement an individual engine.

On turn 15, with distance between apparatuses 75 m, cosmonaut had to return aboard ship, after staying outside him on the whole approximately 2 hour 10 min.

The jet pack for movement in space was the original in individual equipment of American astronaut. The operation of the pack is based on the principle of the ejection of air jet, which is stored under pressure in two tanks (Fig. 53). For the forward movement are intended two nozzles, attached at the ends/leads of the horizontal rod, for the backward movement - one central nozzle. The reserve of air provides thrust 0.9 kgf during 20 s (1 kgf thrust 1 s it makes it possible to move on 0.67 m).

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Cosmonaut must hold in the center of gravity of body and for the displacement/movement in space by narrow pulses depress lever of control. For the rotations with the pressure/clamp to the lever it is necessary to diverge pistol to the side.

During November 1966 took place flight of ship "Gemini-XII". In this flight are realized the rendezvous and mating with the specially

launched rocket "Agena", the output of one of the cosmonauts into open space.

As it is noted in foreign press, all experiments with the ship "Gemini indicated" were conducted for an evaluation of possibilities of man to observe visually ground-based and space objects and adjustments of equipment of photographing such objects; for investigation of the possibility of the egress of man into space for considerable period and determining his portability and to satisfy different operations; for studying conditions for space radio communication for purpose of development of matching systems of connection/communication and finally for conducting of experiments for the reception of television images under conditions of low illumination of objects and, in particular, at night.

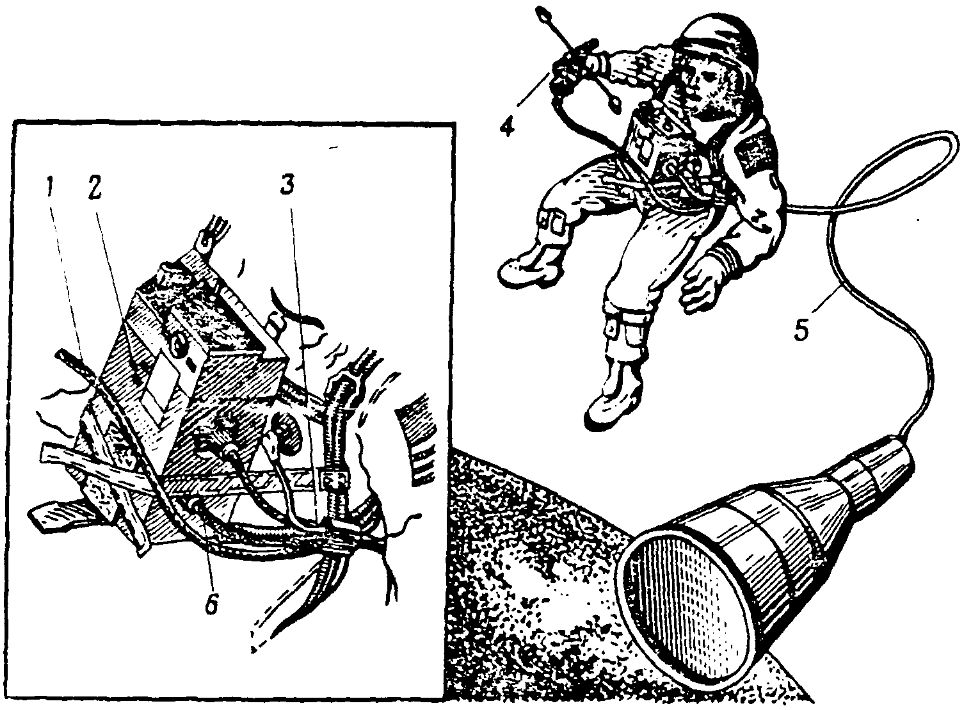

Fig. 53. Breast pack: 1 - pipeline of supply of compressed air to the jet pack; 2 - pack; 3 - electric wire; 4 - jet pack; 5 - halyard; 6 - pipeline of supply of oxygen.

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"Apollo" spacecraft.

At present in the USA there is conducted extensive work on creation of "Apollo" spacecraft for guaranteeing debarkation of people to Moon and their return to the earth. General idea about it in comparison with the already created apparatuses can be obtained from the table, given lower Fig. 54 shows its form.

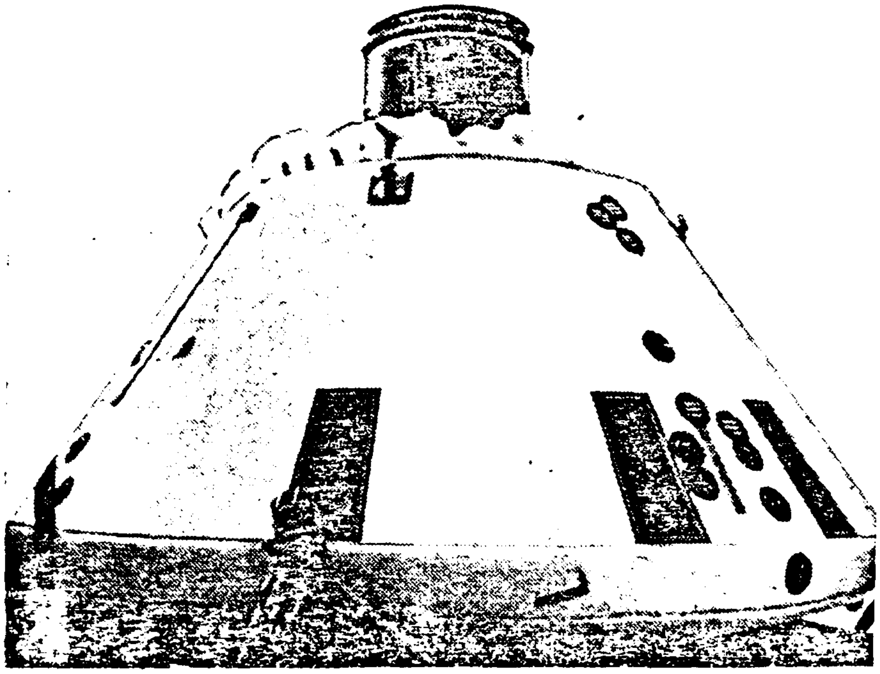

Fig. 54. General view of crew compartment of "Apollo" spacecraft.

Comparative table of the spacecraft of the USA with the crew aboard.

(1) Космический корабль	(2) Наименование данных корабля и ракеты-носителя				
	(3) численность экипажа	(4) объем кабины на одного космонавта, $м^3$	(5) вес корабля, кГ	(6) ракета-носитель	(7) тяга двигателей ракеты-носителя, кГ
(8) „Меркурий"	1	1,56	1600	(9) „Атлас"	160000
(10) „Джеминай"	2	1,13	3200	(11) „Титан-2"	195000
(12) „Аполлон"	3	2,07	42600*	(13) „Сатурн-5	3400000

Key: (1). Spacecraft. (2). Designation of data of ship and carrier rocket. (3). number of crew. (4). space of cabin/compartment for one cosmonaut, m^3. (5). weight of ship, kgf. (6). carrier rocket. (7). thrust of engines of carrier rocket, kgf. (8). "Mercury". (9). "Atlas". (10). "Gemini". (11). "Titan-2". (12) "Apollo".

(13). "Saturn-5".

FOOTNOTE[1]. In the lunar version. ENDFOOTNOTE.

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Fundamental elements of "Apollo" spacecraft are carrier rocket "Saturn-5", strictly spacecraft and lunar cabin/compartment (Fig. 55). Spacecraft must supply three cosmonauts to the low lunar orbit. Then from the ship will be separated/liberated the lunar cabin/compartment, in which two cosmonauts will go down to the surface of the Moon. After staying for a while on it, cosmonauts will return in the lunar cabin/compartment to the low lunar orbit and will be butted on it with the spacecraft, which will supply cosmonauts to the earth. The third cosmonaut is situated always in the spacecraft.

Between "Apollo" spacecraft and carrier rocket "Saturn-5" is placed adapter, which closes lunar cabin/compartment in trajectory phase from launching to the beginning of the docking maneuver. It is a sandwich construction with an aluminum honeycomb filler and weighs approximately 1770 kgf.

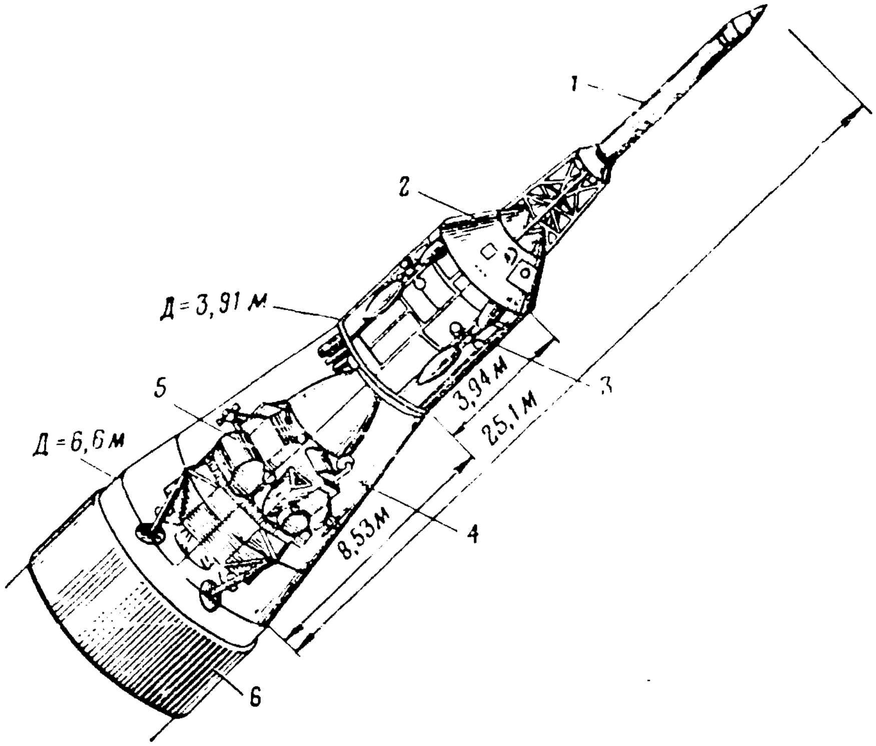

Fig. 55. System "Apollo", assembled on carrier rocket (figure): 1 - system of emergency recovery; 2 - crew compartment; 3 - auxiliary cut off; 4 - adapter; 5 - lunar cabin/compartment; 6 - carrier rocket.

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Adapter consists of two conical sections, connected by the bases/roots (it is lower) the base/root of upper section it is fastened to the upper base/root of lower section). Each section has four identical panels, which are thrown off with the aid of the explosive charges of media by the mating of crew compartment with the lunar cabin/compartment. Lunar cabin/compartment is established/installed directly above the latter/last booster stage. Above it is placed engine it cut off and it cut off the spacecraft crew.

Aboard "Apollo" spacecraft is provided for system of emergency recovery during start and in initial phase of flight. It is installed on the apex of the cone of the crew compartment and is the three-meter truss tubular construction/design of heat-treated titanium alloy. On it there are installed three engines: RDTT with a thrust of 70300 kgf for the department/separation of crew compartment from the carrier rocket; the engine of pitch control and rocket engine for the dropping of farm/truss during the satisfactory completion of the initial track-out phase. The sensor of angles of attack and two opened surfaces, intended with its turn, is located on the end of the truss. Supplementary protective coating protects it cut off crew from the effect of exhaust gases of RDTT of the system of emergency recovery. During the successful starting/launching the system of emergency recovery is thrown off from the ship, as soon as it will achieve safe altitude. With the onset of emergency situation during the start or in the initial phase of flight (to the altitude of 65 km) the system of emergency recovery separates/liberates crew compartment from the carrier rocket and it lands with the aid of the parachute. Flight termination in the case of emergency can be accomplished/realized either automatically (it will operate/actuate the acquisition system of emergency situation on the carrier rocket), or by hand - by cosmonauts. During the detection of serious malfunction any crew member can switch on the system of emergency recovery.

Construction/design of the "Apollo" spacecraft.

Ship consists of crew compartment and power bay.

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Crew compartment, made in the form of cone, is intended for work and recovery of cosmonauts during entire flight to Moon and vice versa (Fig. 56). In it are assembled the airconditioning systems, life support, connection/communication, control of ship, orientation upon the entry and the atmosphere, and also navigation equipment and the on-board digital computer (Fig. 57).

In power plant bay (Fig. 58) there are placed radiators of conditioning system, electric power supply system, tanks with fuel/propellant and compressed gas for fuel pressure feed to the sustainer engine and orientation system engines of ship, and also certain auxiliary equipment. Weight of power plant bay is 10200 kgf.

During flight conditioning system maintains pressure in crew compartment 0.35 kgf/cm². The thermal insulation of crew compartment is provided by the coating with a thickness of 25 mm from the fiberglass. From the outer side the construction/design is covered with special ablation heatproof material, which is connected with the framework. Upon the atmospheric entry ablation heatproof material and fundamental heat shields support the temperature within the crew compartment within limits of 20-38°C.

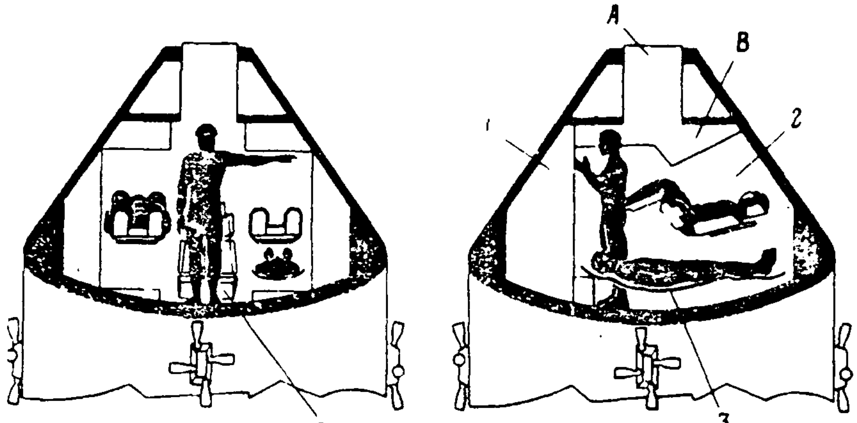

Fig. 56. Crew compartment of "Apollo" spacecraft: 1 - cosmonaut works with navigational instruments; 2 - cosmonaut lies on seat; 3 - cosmonaut lies on cot; A - roof hatch; B - main control panel; C - average/mean seat (folded).

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Despite the fact that at some points of ablative heat shield the critical temperature can reach 2210-2760°C, the temperature of interface between two heat shields is not raised more than 315°C. In this case the temperature of outer skin of the internal housing of crew compartment remains below 93°C.

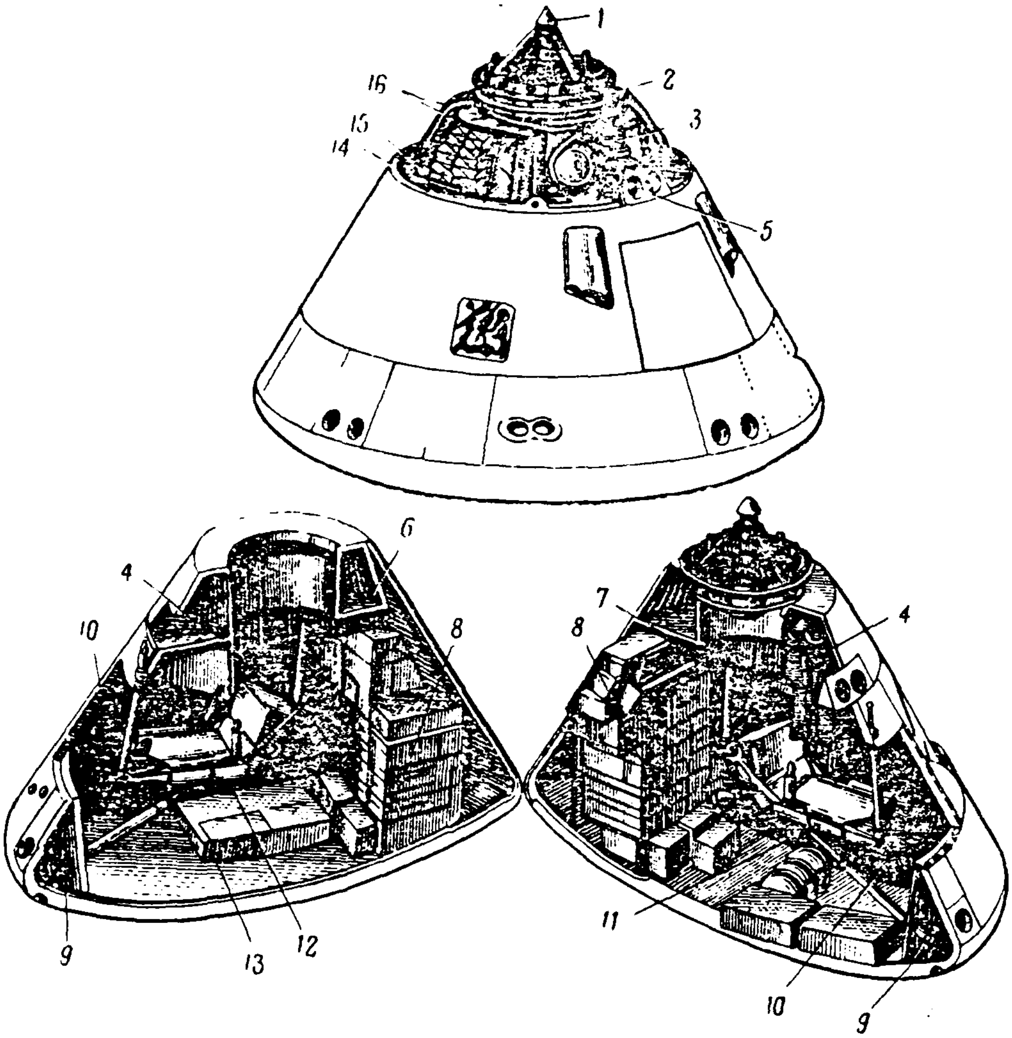

Fig. 57. Arrangement of equipment in crew compartment of "Apollo" spacecraft: 1 - pin, which ensures mating with lunar module; 2 - tunnel for transfer of cosmonauts into lunar module; 3 - grenade launcher for detonation of brake parachute; 4 - upper section of section; 5 - engines for orientation on pitch; 6 - left upper bay for equipment; 7 - right upper bay for equipment; 8, 9 - lower bays for equipment; 10 - flight deck; 11 - bay for equipment under right seat; 12 - bay for equipment under left seat; 13 - lower bay for equipment;

14 - upper bottom of section; 15 - grenade launcher for detonation of drogue chute; 16 - main parachutes.

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From high temperatures, which appear as a result of kinetic heating during orbital injection or as a result of work of orientation system on different flight phases, engine cut off shielded by plug sheet of different thickness. The external panels of power bay as the panel of crew compartment, are composite/compound sandwich construction with the honeycomb aluminum filler. The external walls of four sections (of six that being) are made from aluminum sheets. In them are made the channels, utilized as the cooling surfaces of the radiators of the electric power supply systems and life support. Upper and lower the bottom of power bay they consist of six sectors, which are fastened to the radial beams/gullies. Facing sheets are manufactured from the high-strength aluminum alloy and are linked with the aluminum honeycomb filler, forming composite/compound panels.

In "Apollo" spacecraft are established more than forty high pressure cylinders. They have the most diverse designation/purpose: this containers for the fuel and the oxidizer of engines, for the fuel and the oxidizer of fuel cells and for liquid oxygen of life-support system, pressure bottles (helium), etc.

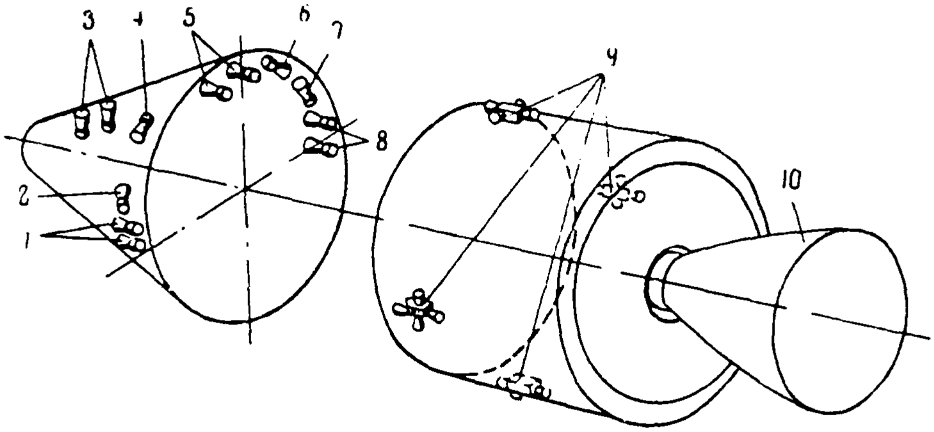

Fig. 58. Arrangement/position of engines of "Apollo" spacecraft: It cut off crew: 1 - engines for orientation on yaw (negative); 2, 6 - engines for orientation along bank (negative); 3 - engines for orientation on pitch (negative); 4, 7 - engines for orientation along bank (positive); 5 - engines for orientation on pitch (positive); 8 - engines for orientation on yaw (positive). Engine cut off: 9 - blocks of pilot engines for the orientation on yaw, bank and pitch; 10 - sustainer engine.

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Walloons are made in the form of cylinders with the spherical bottoms.

Lunar module of "Apollo" spacecraft is intended for delivery/procurement of cosmonauts to Moon from lunar orbit (Fig. 59). This is the first manned space vehicle, designed exclusively for the work under the space conditions.

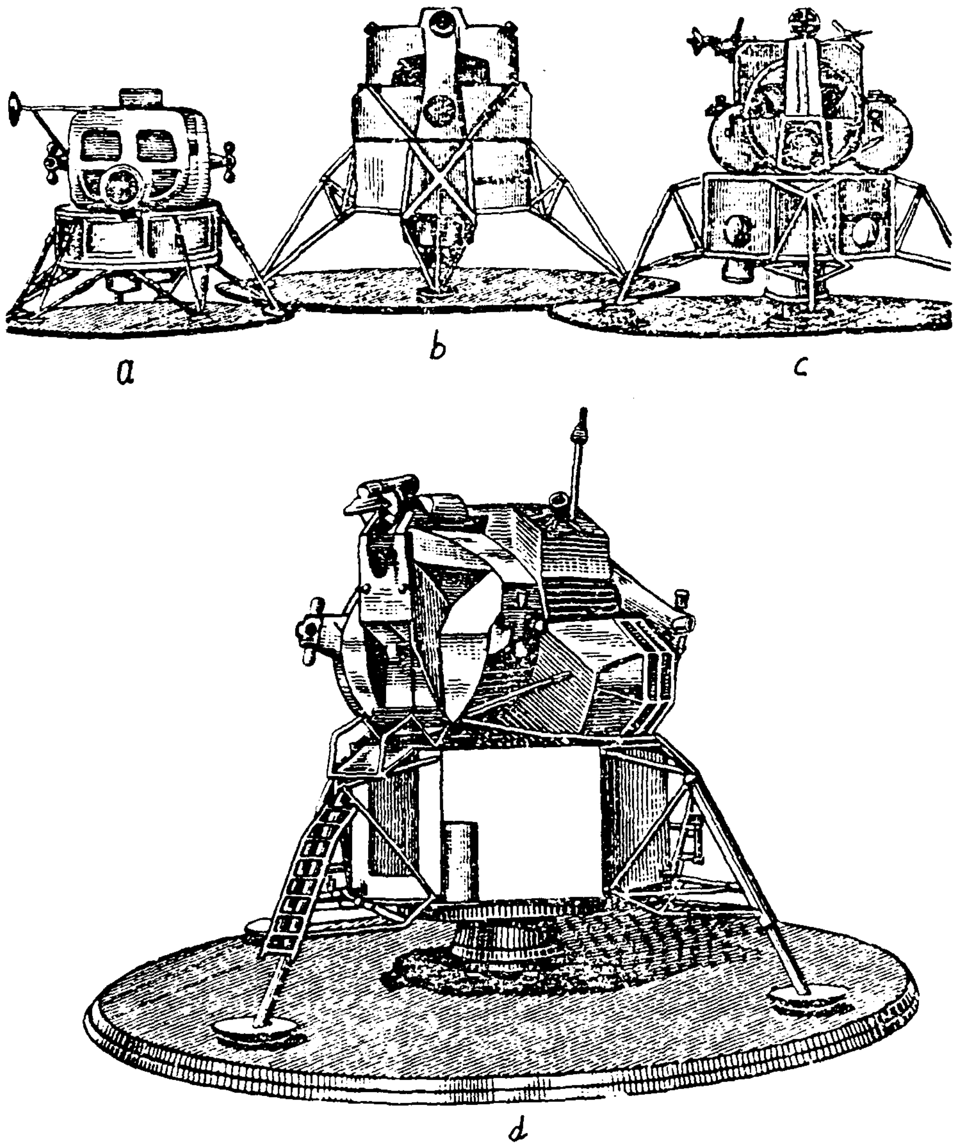

Fig. 59. Models of lunar compartment: a) initial model (1962); b) and c) intermediate models; d) latter/last model.

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Therefore, in its construction/design there was no need for considering aerodynamic factors. Since during the landing of the

lunar module there will be utilized not a parachute system, but a rocket engine, and damping devices of the compartment have a completely different construction/design than the damping devices of the crew compartment aboard the "Apollo" spacecraft.

Lunar module is designed according to this principle: all its parts, not utilized repeatedly, are thrown off. Therefore, the module is divided into the landing and takeoff stages.

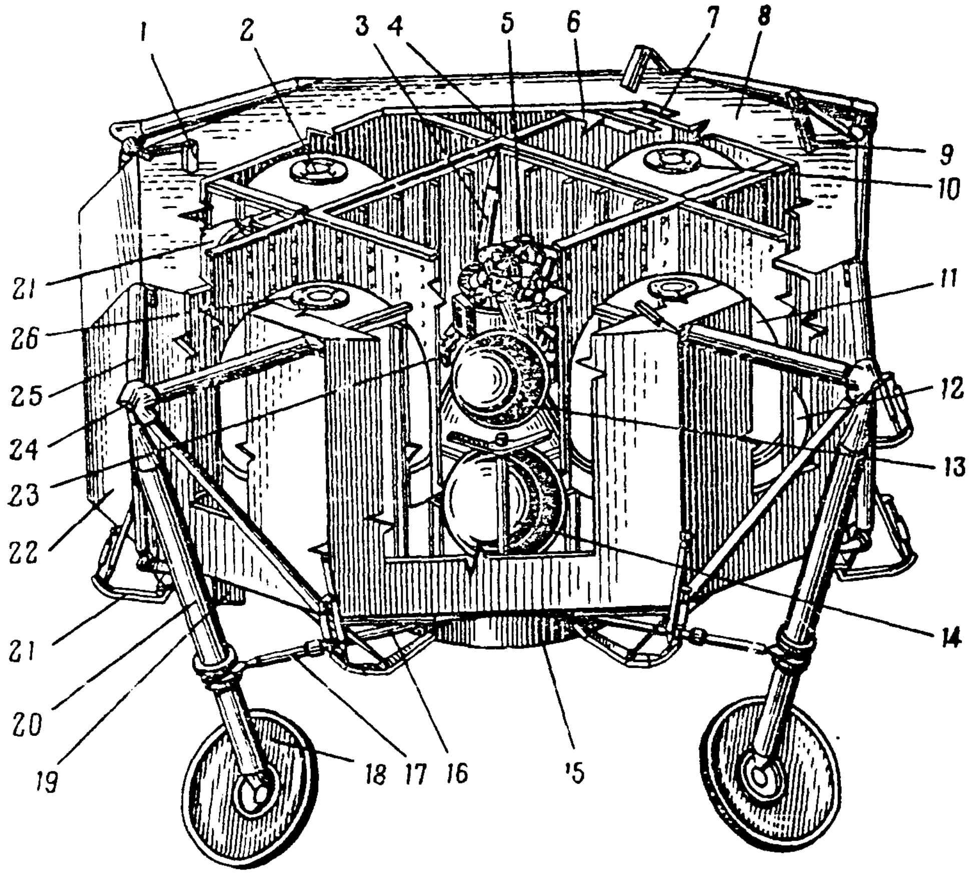

Fig. 60. Landing stage of the lunar module (landing gear in a folded position); 1 - rear device for fastening to takeoff step/stage; 2 - fuel tank; 3 - engine bed; 4 - section for positioning portable life-support system and antenna; 5 - engine; 6 - housing of the stage; 7 - thermal insulation; 8 - heat shield; 9 - front device for fastening to takeoff step/stage; 10 - oxidizer tank; 11 - fuel tank; 12 - section for batteries; 13 - oxygen tank; 14 - tank of liquid helium; 15 - engine cowling; 16, 17 - elements of landing gear; 18 - landing gear strut; 19 - antenna of landing radar; 20 - landing gear strut: 21 - section chassis/landing gear; 22 - section for scientific instruments and equipment; 23 - articulated suspension of engine; 24 -

place of fastening to the lunar module of the adapter between the carrier rocket "Saturn-5" and the "Apollo" spacecraft: 25 - bracket; 26 - oxidizer tank; 27 - small tank with water.

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Landing stage (Fig. 60) serves for the braking and landing of the lunar module and will remain on the Moon. It consists of ZhRD, fuel tanks, equipment for the scientific studies of lunar surface, containers with oxygen, water and helium, and also the landing chassis/landing gear. Chassis/landing gear has four telescopic struts, damping impact loads during the landing, the strut of landing gear of honeycomb construction/design. During the flight in the low lunar orbit the landing gear struts are not produced until crew enters into the lunar module. The engine of landing step/stage is assembled on the gimbal suspension and can be throttled automatically or by hand - by cosmonauts. Thrust can be regulated in the range 480-4800 kgf. The injectors with the variable flow area are utilized for this.

The takeoff stage (Fig. 61) after takeoff and its injection into a low lunar orbit, the mating with the spaceship and the transfer in it of two cosmonauts will be discarded, and it will remain in low lunar orbit. It consists of the cabin/compartment of cosmonauts (Fig. 61), a main ZhRD, steering rocket engines, fuel tanks and has all necessary systems of the flight control, moreover all its controls are duplicated/backed up. Diameter of flight deck 2.3 m, working volume of its 5.3 m^3. Pressure within the cabin/compartment - 20.3 kgf/m^2

(atmosphere it contains 100% of oxygen), temperature of +24°C. The length of entire step/stage is 3.9 m, width at the level of fuel tanks 4.2 m. Framework/body and skin/sheathing of lunar module are prepared from the aluminum alloy, surrounded with a layer of insulating material with a thickness of 50 mm. Main engine of takeoff step/stage is rigidly connected with the housing and provides thrust 1580 kgf. It is intended for the takeoff of lunar module from the surface of the Moon and its removal to the low lunar orbit, but after jettisoning of landing step/stage it can also be utilized, also, for the cessation/discontinuation of landing.

Cosmonauts will control the lunar module standing. As it is indicated in foreign literature, in this position the cosmonaut can withstand the vertical accelerations equal to 8 g, whereas in the process of braking during landing of acceleration 4-5 g must not exceed.

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So that legs of cosmonauts would not slide during takeoff and landing, or under conditions of weightlessness, floor of module of cosmonauts is provided for to cloud by special material. For the support to the upper part of the body in the vertical position are handles and elbow- rests.

In initial version of lunar module on spherical frontal surface of takeoff section it was proposed to make four sufficiently large

windows (for apparatus, which tests/experiences large pressures, spherical form it is ideal from point of view of weight, safety and airtightness).

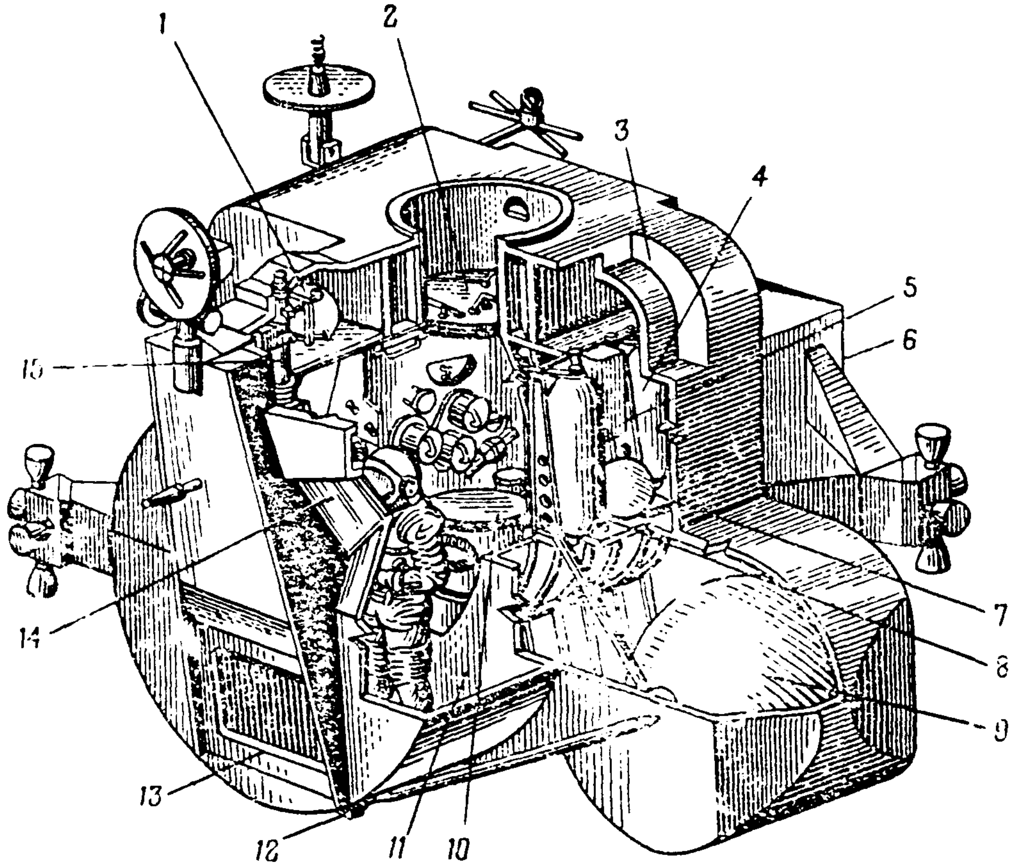

Fig. 61. Ascent stage of the lunar module: 1 - measuring inertial guidance system unit ; 2 - cover of roof hatch; 3 - groove utilized during mating; 4 - fuel tank (for orientation system engines); 5 - reducer of feed system of compressed helium; 6 - assignments section with equipment; 7 - tank with compressed helium (for orientation system engines); 8 - oxidizer tank (for orientation system engines); 9 - fuel tank; 10 - housing of main engine; 11 - module of cosmonauts; 12 - front device for fastening to landing stage; 13 - hatch for entry and egress of cosmonauts; 14 - window of module of cosmonauts; 15 - telescope.

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But calculations showed that during the use of windows of smaller dimensions it is possible to decrease the total weight of lunar lunar module. Therefore it was decided to make two small windows, which nevertheless must provide the required area of survey/coverage. In the final version of window has dual glasses. The clearance between them is communicated with the external medium. Besides the required survey/coverage, glass they provide thermal and radiation shielding.

Butt surfaces are placed on "roof" of lunar module. Hatch for the transfer of crew members into the lunar module and egress from is located here. From the side the lunar module there is a hatch for entry and output of crew members during the stay on the Moon. This arrangement of this hatch removes the need for a difficult and dangerous intrusion to the module. Covers of both hatches are opened inward. Before opening the hatch, pressure in the module must be reduced to the external.

For descent of cosmonauts from ship to surface of Moon it is decided to utilize an aluminum staircase, attached to the front strut of the chassis/landing gear. On the staircase there is a small area/site.

Helium will be utilized as the gas for creation of pressure in fuel tanks of landing stage.

Experiments established that during flight of lunar module jets of hot gas from managers of nozzles of control system can hit the surface of landing step/stage. This necessitated the using of the high-temperature insulation of those places, where the jet can enter. Insulation is made from the refractory metallic plates (several layers).

Before lunar module it will be used for real flight to Moon, it is necessary to test its suitability for execution of this flight. This must be realized as a result of conducting the series of flights along the near earth orbit. One similar flight of the completely equipped lunar module already took place (without a cosmonaut). The cabin/compartment was launched with the aid of the rocket "Saturn-1" together with the mock-up of the "Apollo" spacecraft (Fig. 62). In the process of this testing the performance characteristics of power plant and the operation of the landing and takeoff stages and systems of control were checked.

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The mock-up of crew compartment in this flight completed landing in the ocean. For the landing to the earth the spacecraft has a system of damping impact loads.

During the second flight conditions of crew activity in space will be determined, and also checked is the possibility of the rendezvous and mating of the "Apollo" spacecraft with the lunar module under different ambient conditions. In this flight three cosmonauts will participate. Since the rocket "Saturn-1" (with the increased thrust) can build up relatively small load, aboard the ship there will be placed a limited number of power plants.

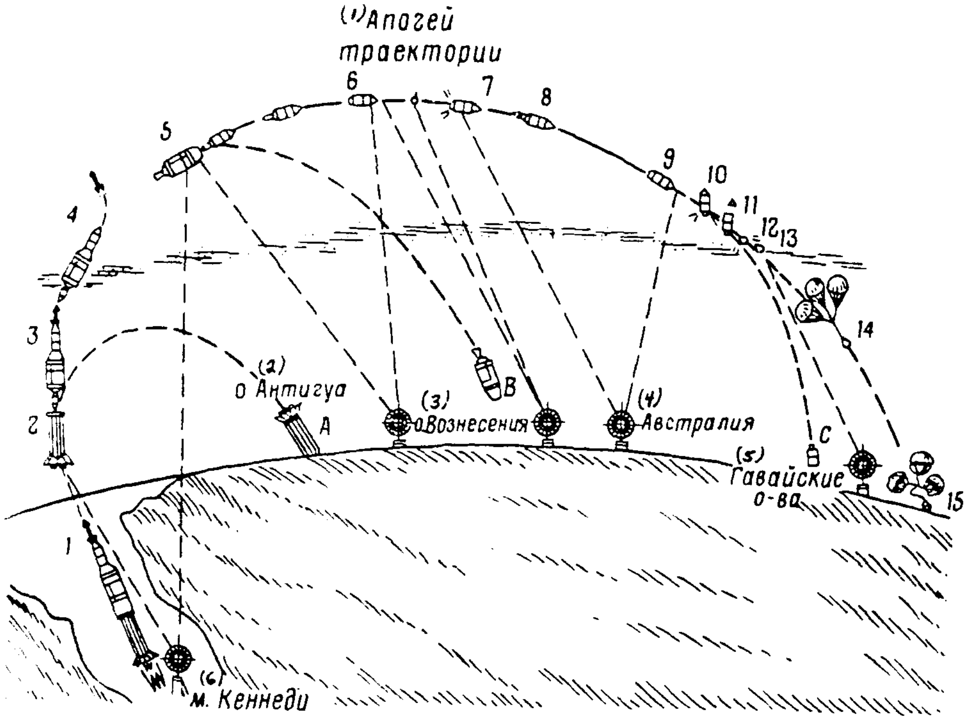

Fig. 62. Mission profile of the carrier rocket "Saturn-1" with mock-up of "Apollo" spacecraft: 1 - missile takeoff; 2 - engine cutoff of first stage and stage separation; 3 - firing second-stage engine; 4 - separation of system of emergency recovery; 5 - engine cutoff of second stage, separation of second stage, first firing of sustainer engine of ship; 6 - cutoff of sustainer engine; 7 - firing of starting motors; 8 - second firing of sustainer engine; 9 - cutoff of sustainer engine; 10 - orientation of ship before separation of sections; 11 - separation of sections; 12, 13 - entry of crew compartment in the atmosphere; 14 - deployment of main parachutes; 15 - splashdown of crew compartment; A - drop of first stage; B - drop of second stage; C - drop of power bay.

Key: (1). Apogee of trajectory. (2). Island of Antigua. (3).

Ascension Is. (4). Australia. (5). Hawaiian Is. (6). Cape Kennedy.

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In foreign press it was reported that on 9 November 1967, there took placed the first launching of the experimental carrier rocket "Saturn-5" AS-501 with "Apollo" spacecraft[1].

FOOTNOTE[1]. Rocket "Saturn-5" will be used also for the real flight to the Moon. ENDFOOTNOTE.

Launching weight of the completely charged/filled carrier rocket together with the payload was equal to 2800 T. Ship consisted of the full-scale basic building block (it cut off crew and engine it cut off) and the mock-up of the lunar module, filled with water.

The launching and flight of the rocket with the ship occurred in this sequence (Fig. 63).

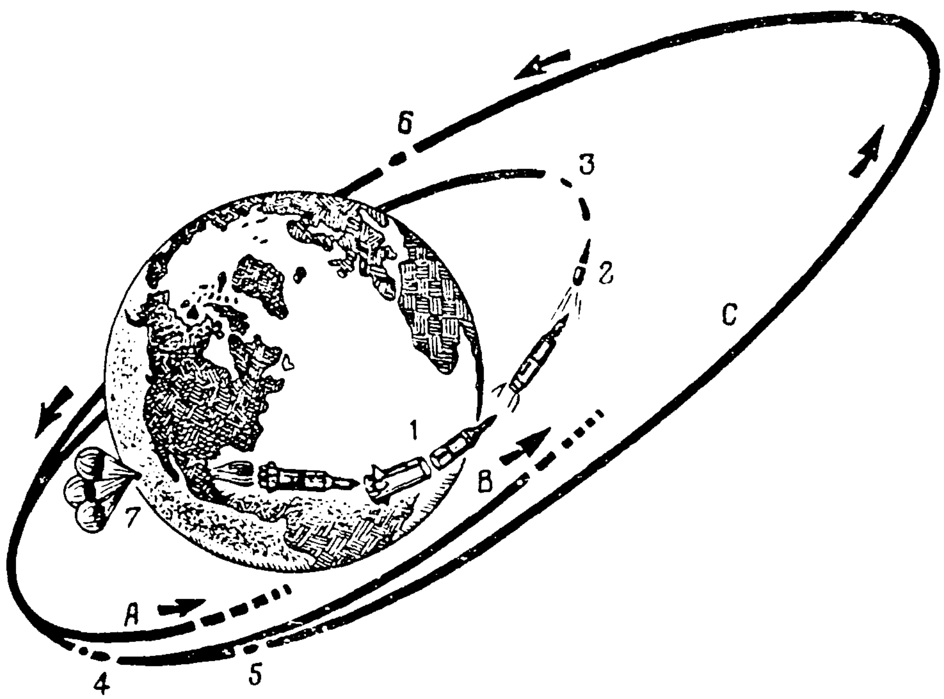

Fig. 63. Mission profile of carrier rocket "Saturn-5" AS-501 with "Apollo" spacecraft: 1 - separation of first stage; 2 - separation of second stage; 3 - ejection of ship in initial circular orbit with altitude of 187 km; 4 - reclosing of third-stage engine, ejection of the ship into an elliptical parking orbit with apogee altitude of 17400 km; 5 - separation of third stage with mock-up of lunar module, first firing of sustainer engine of "Apollo" spacecraft for transfer into a final elliptic orbit; 6 - reclosing of sustainer engine of "Apollo" spacecraft for guaranteeing orbit ejection and entry of crew compartment in the atmosphere with planet escape velocity; 7 - splashdown of crew compartment; A - initial circular orbit; B - elliptical parking orbit; C - final elliptic orbit.

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After breakaway of rocket from starting platform at end of burning of the first stage (after 2 min 30, 72 s from the moment of start) the first stage was isolated from carrier rocket, then were switched on second-stage engines, which, after operating for 6 min 9 s, were switched off, and the second stage was isolated from the carrier rocket.

At the end of burning of the second stage there was primarily set in operation 2 min 20 s the third-stage engine. The third stage of rocket with the "Apollo" spacecraft was put into a circular geocentric orbit at the altitude of 187 km. At the end of the second revolution the third-stage engine, which transferred the stage with the ship to the elliptic orbit with apogee altitude of 17400 km, is repeatedly included/switched on. After the transfer to an elliptic orbit the basic building block of the ship was isolated from the third booster stage and mock-up of lunar module and then the sustainer engine of the ship, which transferred ship to the final elliptic orbit with apogee altitude of 18350 km, was switched on. The sustainer engine of ship was repeatedly switched on after the passage of the ship through the apogee of final elliptic orbit in order to ensure the descent of the ship from the orbit and its entry in the atmosphere with the planet escape velocity - 11074 m/s. Ship completed two "insertions" in the atmosphere. The crew compartment of ship was dropped into Pacific Ocean 1000 km from the Hawaiian islands.

Basic building block and mock-up of lunar module withstood well

conditions for powered flight of carrier rocket. During the orbital flight maximum temperature on the surface of the housing of basic building block, turned to the sun, composed plus of 60°C, on the surface, which is located in the shadow, minus of 73°C. The fuel cells installed aboard the ship functioned normally, liberating water suitable for drinking.

Heat shield of crew compartment satisfactorily underwent tests upon entry into the atmosphere with planet escape velocity. Bottom and lower edge of heat shield underwent the greatest charring; the thickness of the charred layer did not exceed 19 mm. Crew compartment was given in the prescribed/assigned area and was built up onboard the aircraft carrier. The parachute system of the embarkation of crew operated/actuated normally.

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Fundamental tasks of this launching, as the foreign press reports, was the testing of the carrier rocket and "Apollo" spacecraft, heat shield of crew compartment upon entry into the atmosphere with planet escape velocity, and also the testing under heating conditions upon entry into the atmosphere of heatproof seals of the cover of the rapidly opening hatch of the crew compartment.

Further tests of the "Apollo" spacecraft for a complete checking of all characteristics will include both the flights along near earth orbit and output of spacecraft to low lunar orbit (without landing on

Moon). If preliminary tests give satisfactory results, then the decision about the flight of cosmonauts to the Moon will be made.

The "Apollo" spacecraft with crew of three people can ensure duration of flight along geocentric orbit of 10-14 days, along selenocentric orbit of 4-8 days and stay of two cosmonauts on Moon for 24-36 hours.

Parachute of landing system of "Apollo" spacecraft. System is placed in the upper part of the crew compartment of ship. It occupies four sections around the cylindrical air lock. In three sections there are located the main and also drogue chutes and the "guns" of their input into the air flow, into the fourth - two brake parachutes, the "guns" of their input/introduction and the units of the uncoupling of main and brake parachutes.

Conical ribbon parachutes are utilized as brake parachutes. Nominal diameter of each parachute 418 cm, the weight of 10 kgf. Length cord is selected so that parachute canopy would be arranged/located from the heat shield at a distance of the approximately seven diameters of ship.

For purposes of reaching/achievement of stability and smallest impact/shock during deployment drogue chutes have flat/plane slot construction/design. As the bases are used the strip/tape slot parachutes, the diameter of canopy of which approximately 27 m. The

sizes/dimensions of parachutes are selected with this condition so as to ensure rate of descent not more than 10 m/s during the use of two (of three) parachutes.

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In comparison with the previously used parachutes, for spacecraft in the parachute of "Apollo" spacecraft there is increased drag coefficient due to elongation/aspect ratio cord, and aerodynamic interference between separate parachutes is reduced due to introduction to parachute canopy of wide annular slot.

Brake parachutes are introduced in action at altitude of approximately 7000 m. At this altitude operates/wears the pressure-sensor release device, on signal of which is thrown off upper conical cover/cap. After 2 s with the aid of the "guns" into the airflow are introduced two parachutes. By cosmonauts brake parachute can be introduced into the airflow and at higher altitudes than is done with the aid of the pressure-sensor release device. In the stabilized position the ship descends to the altitude about 3000 m, at which on the signal of pressure-sensor release device operate/wear the mechanisms of the uncoupling of brake parachutes and the "gun" of the input/introduction of drogue chutes. Three slot drogue chutes extract the main parachutes. The rate of descent in the ship by the wholly filled parachutes must be 7.6 m/s. The parachutes are disengaged at the moment of contact with the Earth.

In the case of the onset of threat of a serious booster failure at launch or during launching, the landing system operates in following order. The rocket engine of the system of rescue at the start at first is switched on, and in this case the crew compartment is separated from the carrier rocket, and the power bay of spacecraft rises upward to an altitude of 1500 m. Then the rocket engine is turned off, and the two control surfaces, intended for the turn of crew compartment on 180° to the position in the direction of flight, are advanced from the forward section of the framework construction of the starting system of rescue. After the brief period of the flight of crew compartment by the inertia rocket engine, the framework construction of the starting system of rescue and upper lid are separated/liberated from the crew compartment with the aid of the small rocket engines. After 2 s the "guns" introduce into the airflow the brake parachutes, which are opened after 6 s, and the ship descends by wholly filled brake parachutes. Brake parachutes are separated after this, and simultaneously with the aid of the "guns" there are introduced the drogue chutes.

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If it is necessary to interrupt flight at heights/altitudes of more than 7000 m rocket engines of starting system of rescue remain in crew compartment, which completes ballistic trajectory flight. At the altitude of more than 36500 m the position control of ship is accomplished/realized by hand with the aid of jet manager of system. In flight in the dense layers of the atmosphere the aerodynamic

stabilization of crew compartment is provided by the bullet surfaces of the starting system of rescue, which hold its in the position, in which heat shield is located in the forward section. With the reaching of an altitude of about 7000 m, the framework construction of the starting system of rescue and upper lid are thrown off and parachute system is introduced in the action.

In foreign press it is indicated that in process of development of landing system of "Apollo" spacecraft it was necessary to solve number of complicated technical problems. Most serious of them directly or are indirectly connected with the need for the reduction of its weight. The permissible weight of landing system must be equal to approximately 4.4% of landing weight of ship, while the existing parachute systems had a weight about 5.6% of landing weight of ship.

Especially important fact has aerodynamic interference (shading/blanketing), defined by different performance characteristics of separate parachutes in cluster both in that folded, and in filled state. This leads to the fact that the parachutes are filled nonsimultaneously. Furthermore, the nonsimultaneous input/introduction of main parachutes can occur during the specific orientation of crew compartment relative to air flow at the moment of operating the "guns" of the input/introduction of drogue chutes or with the withdrawal of main parachutes from the surface, on which they are packed.

Flight of the "Apollo" spacecraft to the Moon.

Launching. The launching of the "Apollo" spacecraft to the near earth orbit will be accomplished/realized by a carrier rocket "Saturn-5" during the incomplete use of a fuel/propellant of the third stage. From the launcher the carrier rocket must descend after 6-per-second delays on the starting platform. In this period the thrust of the engine installation of first stage will increase to the nominal value.

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Initial acceleration, however, will be comparatively small, since thrust-to-weight ratio will compose only 1.25. After descent from the launcher the carrier rocket will fulfill the maneuver, which will ensure with it the injection to the prescribed/assigned azimuth. At the end of the vertical section of lift, which lasts by 12 s, the program on the pitch will begin automatically to be mastered (up to the end of burning of first stage). The cutoff of central first-stage engine will occur at t+155 s, and peripheral engines - at t+159 s (where t - launch time of rocket from the launcher). By this time spacecraft will be located at the altitude of 60 km and at a distance of 120 km from the place of start.

All operations on carrier rocket will be satisfied automatically. Crew will be occupied mainly with the control of starting/launching and with the exchange of information with the Earth. In this case the

cosmonauts must be ready to the use of a system of emergency recovery, which will ensure with them the possibility of safe descent in the case of emergency in the launching phase, including on the starting platform. Any malfunction in the spacecraft will be received by the acquisition system of the emergency situation, which puts out information to crew. If necessary the cosmonauts can discontinue flight. The possibility of automatic flight termination will be preserved virtually to the end/lead of the work of first stage. This will make it possible to consider the appearance of any dangerous situation to take the appropriate measures.

After separation of first stage will be switched on engine installation of second stage, which achieves complete thrust in t+163 s. In the powered phase of the second stage in the case of the malfunction of the inertial system for control the cosmonauts can take control for themselves, utilizing a guidance system of ship. At approximately t+188 s there will be separated the emergency system of rescue, since from this point on, the inherent engine of ship can separate/liberate it from the carrier rocket. The engine installation of the second stage will operate for approximately 75 s. After this time the speed of rocket will be 6.9 km/s, flight altitude will correspond to the altitude of intermediate geocentric orbit (135 km), and the distance from the place of launching will be 1670 km.

After the cessation of operation of the engine installation of

the second stage and its separation, the engine plant of third stage, which will increase the speed of the rocket to 7.8 km/s, will be switched on. This phase of flight will last approximately by 150 s. The third stage will put the ship into intermediate circular geocentric orbit with a altitude of 185 km at a distance of 2780 km from the place of the launch. Removal in this orbit of payload with the weight of 136,078 kgf will engage approximately 11.5 min and will require approximately 2,494,800 kgf of fuel/propellant. Tracking the rocket in the powered phase of the third stage they will accomplish/realize the specially equipped ships, since during this period the rocket will be found out of the visibility ranges of ground stations.

In parking orbit ship can it completed to three turns, each with duration to 1.5 hour although according to plan/layout third stage with ship it must pass from parking orbit to flight trajectory to Moon on second turn. During this time of flight the crew must test all systems of spacecraft and prepare to the following stage of flight, supporting periodic contact with the ground stations.

Transition/transfer to flight trajectory to Moon (Fig. 64). Flight clearance to the Moon comes from flight control center from the Earth not later than in 15 min until the estimated time of transition/transfer. After obtaining the command, the crew must orient spacecraft and produce the starting/launching of the engine installation of the third stage. It will operate for only 5.5 min.

For this time the speed of the ship will be increased by more than 3 km/s, and the ship will emerge into the flight trajectory to the Moon. The trajectory elements will be determined after this from the Earth, and the precision/accuracy of the injection will be estimated. Then cosmonauts, utilizing manual control and orientation system of power bay, will separate/liberate it cut off crew and engine cut off from the latter/last booster stage, they will dump the adapter of lunar module, will separate it from the ship and is turned on 180° power bay together with the crew compartment. Then with the aid of the pin and the subtending device/equipment they will carry out mating of the forward section of the crew compartment with the lunar module (subtending device, after there is provided the initial contact, draws both parts up to each other and rigidly links them).

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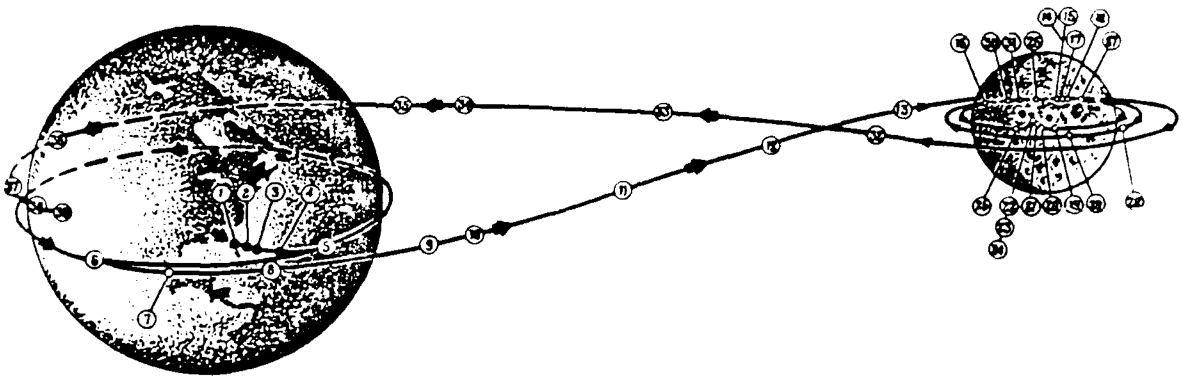

Fig. 64. Flight trajectory of "Apollo" spacecraft for the delivery of the cosmonauts to the Moon: 1 - moment of launch; 2 - activation of power plant of second stage; 3 - separation of starting system of rescue; 4 - activation of power plant of third stage; 5 - exit point in geocentric orbit; 6 - exit onto a trajectory of flight to Moon; 7 - beginning of power-off flight; 8 - beginning of separation and reorganization; 9 - rejoining lunar module; 10 - jettisoning stage; 11, 13 - correction in middle phase of flight; 14, 15 - ejection into a selenocentric orbit and power-off flight; 16 - separation of lunar module and of basic building block; 17 - output of lunar module to transfer trajectory; 18 - power-off flight (before switching on of engine of landing stage of lunar module); 19 - beginning of the work of the engine of landing stage for guaranteeing the braking; 20 - beginning of the stage of final removal into the landing region; 21 - beginning of the stage of landing; 22, 23, 24 - touchdown, the apron of lunar module on the surface of the Moon; the start of takeoff

stage; 25 - power-off flight to the rendezvous with the basic building block; 26 - beginning of rendezvous; 27 - beginning of mating; 28 - rigid docking, the beginning of power-off flight along the selenocentric orbit; 29 - jettisoning lunar module; 30 - beginning of transition/transfer to the flight trajectory to the Earth; 31 - power-off flight along the flight trajectory to the Earth; 32, 33, 34 - correction in the middle phase of flight; 35 - separation of power bay; 36 - entry of crew compartment in the atmosphere; 37 - beginning of descent on brake parachutes; 38 - separation of brake parachutes, the opening of main parachutes; 39 - touchdown.

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Power-off flight on the trajectory to the Moon will be continued more than three days. During this period with the aid of the march engine installation it will be produced to three corrections in the middle phase of flight to the Moon.

Ejection into the selenocentric (lunar) orbit. In selenocentric orbit the spacecraft will leave above the reverse side of the Moon, out of the visibility of terrestrial stations. From the specific distance to the Moon it will begin braking with the aid of the sustainer engine (for this maneuver there is required 11.3 T of fuel/propellant). This will make possible to the gravitational forces of the Moon "to take" ship. Ship will leave in circular selenocentric orbit at a distance from the Moon approximately 150 km. If the maneuver according to the injection of ship into lunar orbit will not

succeed for any reason, then the ship, continuing the power-off flight along its trajectory, will fly around the Moon and will return to the Earth.

In lunar orbit spacecraft will make three turns around Moon during 5.5 hour. For this time ground stations will accurately determine its orbital parameters, and the crew will test the orientation system of lunar module and will withstand into it the equipment, necessary on the surface of the Moon. Two cosmonauts, which must go down to the Moon, will pass into the lunar module. Then cosmonauts will close hatches and will begin preparation/training for the separation of lunar module from the spacecraft.

From "Apollo" spacecraft lunar module will be separated upon switching on for 5 s the engines of the lunar module. After this, two flight vehicles slowly will be separated from each other. When the distance between the module and the ship achieves 18 m, cosmonauts in the lunar module will scan by its in such a way that the cosmonaut who remained aboard the spacecraft could inspect it and be convinced of the fact that it has no external damages. Meanwhile, in the lunar module cosmonauts will finally test controls and instruments and will prepare module for the conclusion/output to the intermediate orbit. Lowering lunar module will be begun, when ship will be located above the invisible side of the Moon. The engine of landing stage will be turned on for 52 s. The lunar module will pass to the gradually descending trajectory.

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The crew of lunar module during the descent with the aid of onboard RLS [radar] will follow the spacecraft and determine the parameters of its line of descent.

The cosmonaut who remained in spacecraft with the aid of instruments will follow the lunar module, being oriented along its indicator lamp. When the lunar module enters into the visibility range from the Earth, after its motion for refining the navigational information ground stations can follow.

If will accepted solution not produce landing lunar module, it will remain in safe orbit of expectation, with which it will be able to begin maneuver for rendezvous with spacecraft, or spacecraft will pass in orbit of lunar module for rescue of its crew.

Descent of lunar module from selenocentric orbit to surface of Moon will occur into three stages: braking, removal in landing region and strictly landing.

Stage of braking will be prolonged by almost 8 min, the module for this time will pass 400 km, will be lowered to an altitude of 26 km above the Moon and will enter into the landing region at a distance of up to 15 km from the selected point. Braking is provided with the aid of the engine of landing system, which works on the complete thrust. The flight trajectory of module in the operating cycle of

engine is almost horizontal.

Stage of removal of lunar module into landing region will be begun from its turn, accomplished so that cosmonauts could observe selected area. In this stage the engine of landing stage gives 60% of complete thrust and in less than 1.5 min will lower the flight speed of the module from 137 to 15.2 m/s. At the end of the stage the module will be dropped/omitted to the altitude of 150 m above the surface of the Moon approximately 360 m of the selected landing place.

In stage of landing flight will completely control/guide cosmonauts. They will ensure the vertical alignment/levelling of lunar module, the gradual decrease of the engine thrust and vertical descent from the altitude of 30 m. The minimum duration of landing is 75 s; however, this stage can last longer if the specific time to the inspection of the landing region is required by the cosmonauts.

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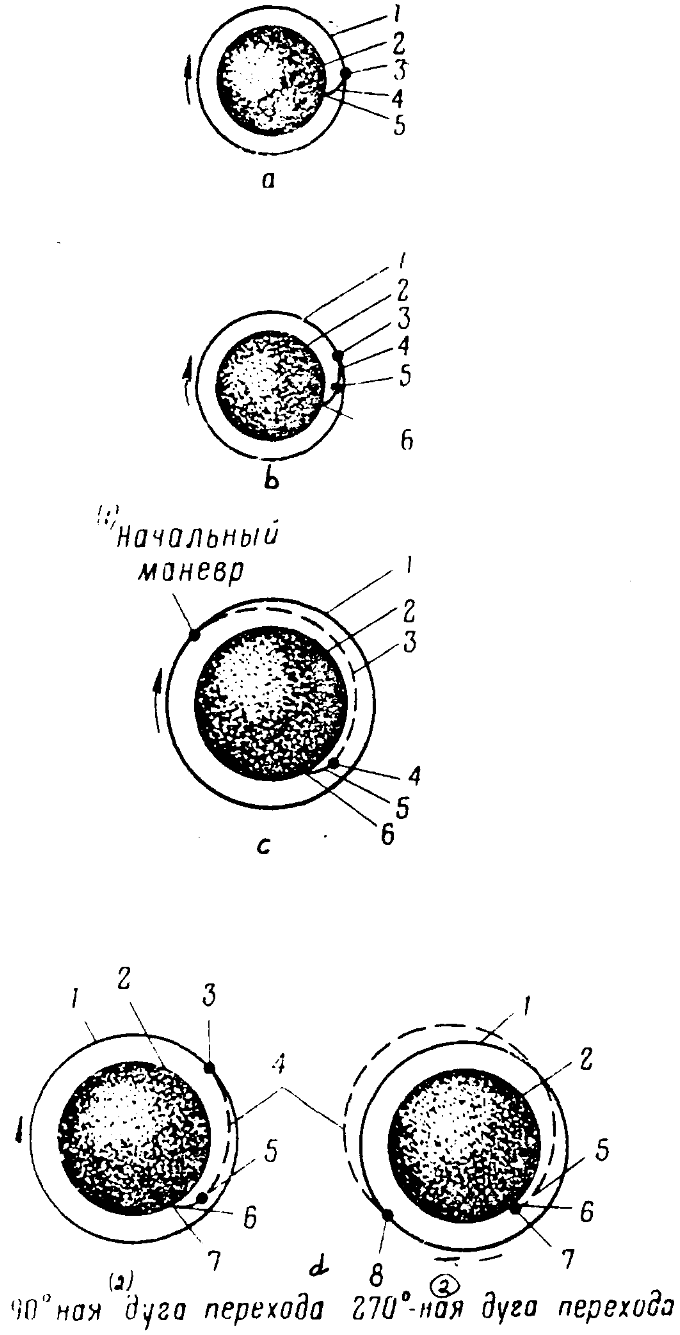

Fig. 65. Trajectories of descent of "Apollo" spacecraft to Moon: a) descent with continuous constant thrust: 1 - low lunar orbit; 2 - surface of Moon; 3 - point of firing engine; 4 - trajectory of descent; 5 - landing place; b) descent with continuous

variable/alternating thrust: 1 - low lunar orbit; 2 - surface of Moon; 3 - point of firing engine; 4 - trajectory of descent; 5 - transition point from minimum thrust to maximum; 6 - landing place; c) descent along ellipse: 1 - low lunar orbit; 2 - surface of Moon; 3 - orbit of free flight; 4 - point of firing engine; 5 - maneuver with descent; 6 - landing place; d) descent along ellipse (synchronous): 1 - low lunar orbit; 2 - surface of Moon; 3 - initial maneuver; 4 - orbit of free flight; 5 - point of firing engine; 6 - maneuver with the descent; 7 - landing place; 8 - initial maneuver.

Key: (1). Initial maneuver. (2). ... arc of transition/transfer.

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Fig. 65 gives different trajectories of descent of "Apollo" spacecraft to Moon, while Fig. 66 shows ship before landing on Moon.

Surface stay of Moon. After landing the cosmonauts will rate/estimate situation aboard the ship and will test efficiency of all systems of lunar module. Cosmonauts must ascertain that there is no need for premature return to the spacecraft. Then they will supply to the safety device the igniter of the engine of landing stage, and fuel vapors will be released from the tanks of this stage, they will install the gyroscope-stabilized platform of guidance system and will disconnect all systems, which must not operate in the period of stay on the Moon. During the subsequent period, when ground stations will more precisely formulate the coordinates of landing place, cosmonauts will test the apparatuses, intended for the determination out of the

lunar module: pressure suits with the liquid cooling, pressure suits, heatproof and antimeteorite clothing, special foot-wear for the movement by the lunar surface, the glove and supplementary shielding reflectors on the helmets.

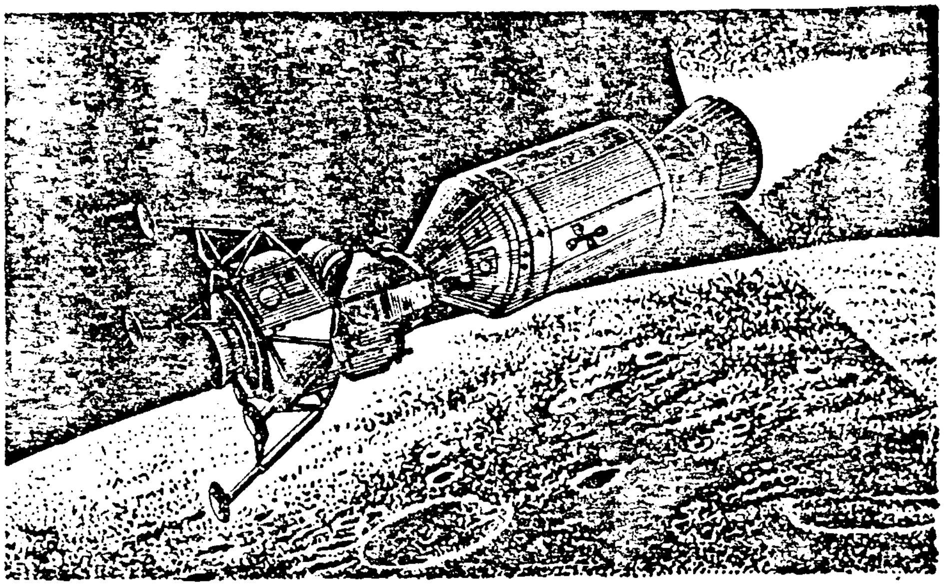

Fig. 66. "Apollo "spacecraft before landing on Moon (figure).

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Then each cosmonaut will put on the individual portable life-support system, after which the cosmonauts unseal the lunar module and go out to the surface of the Moon. All operations prior to the egress will last approximately 1 hour, 45 min.

An 18-hour stay of the cosmonauts on the surface of the Moon is planned. For this time each cosmonaut will conduct two cycles of investigations with a duration of up to 3 hours, i.e., as a whole on the surface of the Moon there will be carried out work in the space of 12 man-hours.

With the first egress one of the cosmonauts will inspect the

lunar module and the depth of the trace left by the landing gear struts in the soil, will record other technical specifications which can prove useful for future landings. Meanwhile, the second cosmonaut will transmit information about the lunar surface to the earth and photograph the surrounding objects. Then the cosmonauts will remove equipment from the lunar module and will conduct some experiments, and they will also install on the surface of the Moon a parabolic high gain antenna, after which they will transmit to the earth television images.

One of fundamental tasks which will face the cosmonauts in this flight will be the gathering of lunar samples. Cosmonauts will fill one of the containers on them and will immediately deliver it into the lunar module in order to have these samples even in such a case when any unforeseen facts prevent their further collection. If after conducting of the enumerated operations it remains sufficiently time and cosmonauts greatly will not get tired, they will install on the surface of the Moon scientific instruments.

After finishing investigations, both cosmonauts will return into lunar module, hermetically seal it, remove their protective clothing and take food. One of the portable life- support systems will be recharged for the eating period. Then there will follow a 6-hour sleep period for the cosmonauts, for th time of which there will be recharged the second portable life- support system. After sleep and a second eating of food, the cosmonauts will put on the protective

clothing and will prepare for the second cycle of investigations, during which they had to gather samples from sections which are found at the sufficiently great distance from the lunar module and install there scientific instruments.

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On completion of the second cycle of operations the cosmonauts will return into lunar module, they will test installed equipment and will prepare takeoff module for start from Moon.

In retention time of lunar module on surface of Moon cosmonaut, who remained aboard spacecraft in crew compartment, periodically checks all ship systems and supports direct connection/communication with Earth and indirect (through terrestrial relay stations) - with lunar module.

Start of ascent stage of the lunar module from the Moon. After completing all pre-firing checks, cosmonauts will await, when spacecraft with the third cosmonaut passes on the next turn above the landing place of lunar module and angular distance between it and module will achieve 9°; in they will include the engine of takeoff stage.

With start of the ascent stage of the lunar module explosive charges will break mechanisms and lines which link it with landing stage. During first 12 s after start from the surface of the Moon the ascent stage of the lunar module will be built up vertically, and then the onboard guidance system will bring it out at the optimum altitude for the achievement of orbital speed. Trajectory of climb will end at the altitude of 15 km. The engine of takeoff stage will report this speed to the lunar module, which will make possible "to support" the

orbit, which is ellipse with a smallest distance from the Moon 15 km and with the greatest - 55 km. During the rotation along this orbit takeoff stage will not fall to the Moon, even if as a result of any malfunction it will be impossible to complete the provided for maneuvers for the rendezvous with the "Apollo" spacecraft. With the normal course of flight takeoff stage as a result of a number of narrow pulses of the reactive/jet orientation system engines must carry out rendezvous and mating with the ship approximately 2 hours after start from the Moon. The butted "Apollo" spacecraft can remain in orbit of rendezvous how much it will be necessary.

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After the mating of lunar module with the ship the cosmonauts with all materials and the samples, assembled on the surface of the Moon, will pass from lunar module to the crew compartment of ship, then they will separate the lunar module from the ship and will begin preparation/training for the translation/conversion of ship from the selenocentric orbit to the flight trajectory to the Earth.

Transfer from the selenocentric orbit to flight trajectory to Earth. This maneuver will be accomplished/realized above the invisible side of the Moon, when spacecraft is not sunlit. Soon the sustainer engine will begin to work after and spacecraft again will enter in the field of the vision of the ground servo stations, will be begun continuous tracking the ship. For the temperature balance of the surface to the ship, there will be given a slow long running

relative to the longitudinal axis, after which the cosmonauts will lie down to sleep. The ship will follow on the computed heading, which ensures the safe conditions for entry into the atmosphere and the moment of entry, which makes it possible to carry out a splashdown in the main landing region. For the satisfaction of these conditions, the correction of motion in the middle trajectory phase can be required. The power bay will be discarded approximately 15 min prior to the atmospheric entry. Entry into the atmosphere will be begun at an altitude of 120 km. The distance between the entrance point and the point of splashdown must be 2400-4000 km. The front heat shield will be discarded at an altitude of about 7 km above the surface of the Earth, and there will be opened two brake parachutes, intended for the orientation capsules to the alert status for disclosing/expanding the main parachute, and also for a reduction in speed from 120 to 60 m/s (at an altitude of 3 km). Then the three main parachutes will be opened with the aid of the special drogue chutes. Rate of descent decreases to 7.6 m/s.

Prospects of applying the spacecraft of the type "Apollo".

American specialists assume that developed for flight service of cosmonauts to Moon carrier rocket and "Apollo" spacecraft have great potential capabilities for solving series of problems, which will be examined after implementation of program, which foresees flight of spacecraft with crew of three people to Moon and his return to the earth.

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In particular, they want with the aid of these ships to set up in rheocentric orbit the manned station with the large effective life, and also lunar bases. They plan to utilize these ships, also, for the flights to other planets.

For next years it is proposed to carry out sixteen flights of manned spacecraft along geocentric orbits, nine flights along selenocentric orbits and landings on Moon. For studying different questions of space flight aboard each ship a large quantity of different research equipment will be established/installed. But the ship must have strictly defined weight (depending on the purpose - 10-48 T). The necessary weight they think to achieve by changing the fuel reserve for the near-earth and circumlunar flights. For an increase in the duration of flight are intended to increase the period of the action of the primary systems of the apparatus such as energy, communications, guidances, and life supports.

Research equipment will be placed in all sections of the spacecraft. In the ascent stage of the lunar module, intended for the landing on the Moon, a cabin-laboratory will be equipped. Crew compartment (command) and auxiliary section can be isolated from the ship and freely maneuver in space during the specific time, and then again be drawn together with it and be mated for continuing the flight. The engine of the auxiliary section of the apparatus, which develops thrust of about 10 T, will ensure the increase in the speed,

sufficient for changing the angle of the orbit inclination or 60°.

For flights around Earth spacecraft it is planned to launch into orbit by altitude from 160 to 370 km with slope angle toward equator from 0 to 90°. If flights will be accomplished/realized for purposes of astronomical observations, ship will be equipped with optical telescope with the appropriate filters and photographic equipment.

In the same layout it is proposed to launch the vehicle with crew of three people on steady-state orbit altitude of 31000 km. Duration of flight in this orbit there will be 45 days. All sections of ship for this flight will be modified.

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Inertial guidance system will ensure the necessary position of ship with an accuracy to ±5°.

Flights along selenocentric orbits will be accomplished/realized for investigation of circumlunar conditions, obtaining of image of lunar surface and study of its characteristics. Minimum flight altitude above the surface of the Moon of 40 km is planned. For such flights the spacecraft will be equipped with the system of the wide-angle chambers/cameras, which make it possible photograph various small areas large areas, and also ensuring multispectral photography with the high resolution.

Foreign specialists discuss possibility of applying probes for studying almost inaccessible sections of lunar surface. In the probes, which it is proposed to throw off from the spacecraft to the surface of the Moon, will be placed television cameras, instruments for the geophysical measurements and equipment for conducting different experiments (Fig. 67).

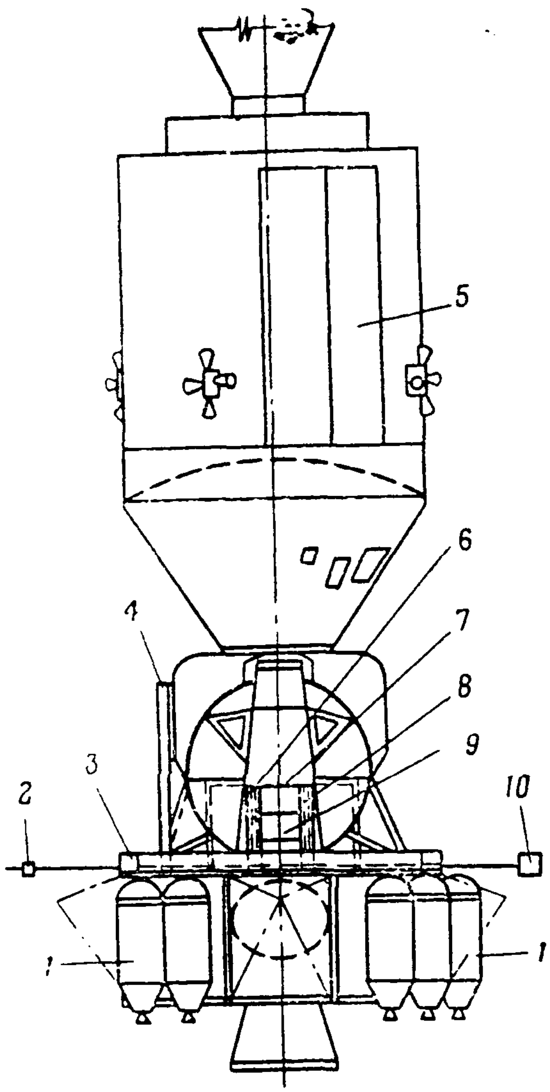

Fig. 67. Space vehicle ejected into selenocentric of orbit: 1 - probes for study of lunar surface; 2 - magnetometer; 3 - equipment for radar mapping; 4 - collector of micrometeorites; 5 - system of cameras for the mapping and survey/coverage of the surface; 6 - multispectral camera; 7 - spectrometer of ultraviolet range; 8 - sensor of infrared radiation; 9 - radio altimeter; 10 - gravimeter.

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The spacecraft intended for near-earth and circumlunar flights, in essence, will be identical, only the power plant of the landing stage of the lunar module will be preserved in ship for circumlunar flights. If necessary, when, for example, it is necessary to change the plane of orbit of the ship, it will ensure augmented thrust to the thrust of primary power plant or it will be utilized as a standby power plant for the ejection of the ship into the trajectory of return to Earth.

In the construction/design of spacecraft for flight to Moon it is proposed to introduce serious changes mainly for purposes of increase in period of action of different systems. Furthermore, is provided for the development of pilotless transport section and the resolution of the problem of the guarantee of landing this section on the Moon.

It is proposed to accomplish/realize landings on the Moon at any point of its surface. The stay of two cosmonauts on the Moon of 14 days is planned. The spacecraft, which will supply to the Moon the pilotless transport section (instead of the takeoff stage) with the shielding shelter for the cosmonauts, the research equipment, food and other reserves, will be first launched. The weight of the entire load of transport section will be 2.7 T. Then another spacecraft will supply into the same point of the Moon lunar module with two cosmonauts aboard. After landing on the moon, cosmonauts will pass into the previously supplied shielding shelter for conducting of the outlined investigations. In the lunar module they will return after

completion of the program of experiments. Utilizing the ascent stage of the lunar module, cosmonauts will leave the Moon, after rendezvous in selenocentric orbit with the spacecraft and mating with it they will pass in it and will return to the earth.

Speaking about the more distant futures of using a spacecraft of the type "Apollo", it is possible to indicate the striving in the USA, after completion of the basic part of works connected with manned flight to Moon, to begin the prelaunch activity to Mars.

Most favorable periods for realization of flight for Mars will be 1971 - year of closest approach, 1973 and 1975.

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After 1975 Mars will be removed from the Earth, the activity of solar bursts at the same time will be raised. Therefore for the realization of flight for Mars it will be necessary to utilize a more powerful carrier rocket and to considerably increase the weight of ship, since it is necessary to install more massive protective shields.

In preliminary development duration of flight of manned spacecraft to Mars is taken as equal to 390-440 days (is here taken into consideration and period of stay on Mars - from 10 to 40 days). For the starting/launching of spacecraft to Mars it is proposed to utilize carrier rockets with ZhRD, which operate on a high-energy chemical propellant. This will make it possible to carry out flight

considerably more rapid than in the case of applying the carrier rockets with the nuclear engine. To assemble spacecraft and all necessary booster stages is planned not on the Earth, but in the earth's orbit.

On available projects, spacecraft for flight in Mars is construction/design of triangular planform with biconvex profile of great relative thickness (Fig. 68). The basic part of the triangle, formed by three sections of ship, occupies that not reversed to the earth it cut off. It is a truncated cone with elliptical bases. Weight of it is 14 T. In it are placed living quarters of crew, the laboratory and the command center of ship.

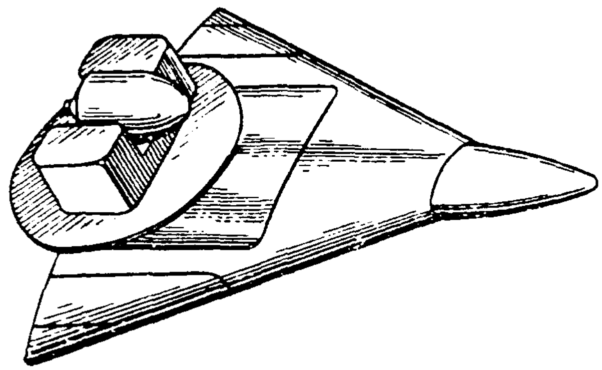

Fig. 68. Manned space vehicle for flight to Mars (figure).

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In the geometric center of the large base/root of section the main power plant of ship is placed. It will be used for the communication/report to ship of planet escape velocity for the return from Mars and the stabilization of ship in orbit around Mars. For the creation of artificial gravitational force the power plant will be advanced on the telescopic rods and impart rotation to ship. To the upper part of this section is connected that returned to the earth it cut off with the flight deck and all systems, necessary for guaranteeing entry into the Earth's atmosphere and landing in the assigned area.

On upper surface of irreversible section will be established/installed also it cut off, intended for landing on Mars. According to initial project, it had to be the combination of disk-shaped heat shield and rectangular laboratory with the deepening in the center section for positioning/arranging the takeoff stage.

The maximum weight of this section must not exceed 25 T. In the section there will be installed two fundamental ZhRD - landing and takeoff. They must provide multiplying. The thrust vector control will be accomplished/realized by a deviation of the nozzle, for which it is proposed to install engines on the hinged suspensions. For orienting the section out of the atmosphere of Mars will be established/installed the engines of the system of the position control, which operate on the same fuel as primary power plants. In all cases this section must consist of the command section, the section of maintenance/servicing and section with the habitable and working locations. The section with the habitable and working locations, located in the rear end of the section, must have four departments/separations - laboratory, toilet, kitchens and locations for recovery.

Total calculated weight of spacecraft is approximately 48 T, and its crew consists of six people.

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Sequence of flight (Fig. 69) assumes;

a) assembly in earth's orbit of manned ship with booster stages:

b) launching of the ship from earth's orbit and flight to planet Mars;

c) rocket braking for removal of the ship in a circular orbit around Mars and then braking due to resistance of upper air of Mars;

d) separation from ship of section, intended for landing on Mars,

descent and its landing on surface of planet;

e) return of cosmonauts to the earth after stay on Mars.

Selection of area/site for landing on Mars is produced from the circular orbit of Mars. Then the three cosmonauts pass into the section, intended for the landing, separate it from the spacecraft and in an 1 hour are lowered to the surface of Mars. With the orbit ejection the engines of the system of the position control are utilized for the separation of section and its orientation. Orbit ejection provides the fundamental landing engine of section. entry into the atmosphere of Mars begins approximately 28 min after orbit ejection, at an altitude about 150 km.

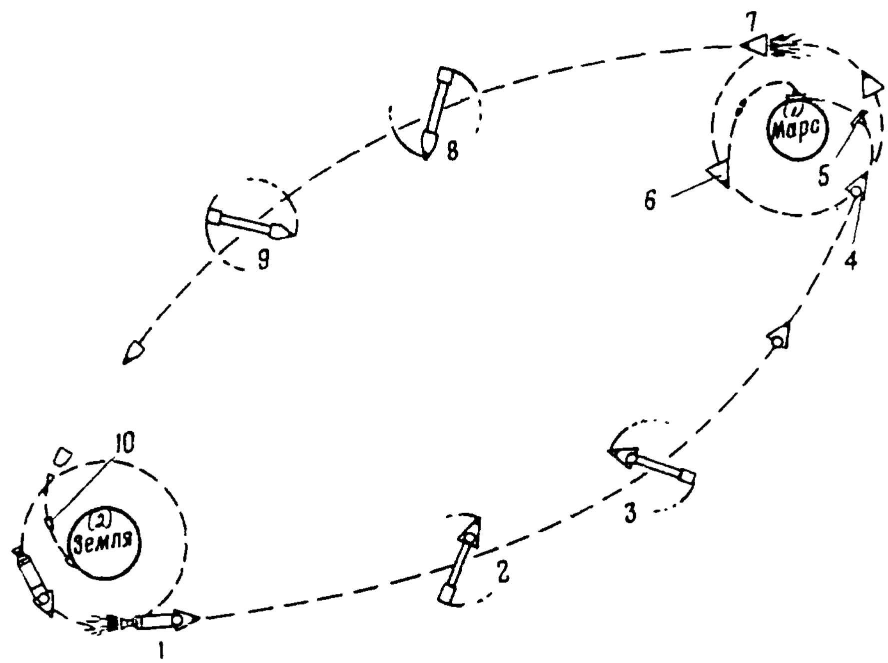

Fig. 69. Sequence of flight to Mars and back: 1 - start from orbit of Earth; 2, 3 - rotation of spacecraft for braking by engines; 4 - entry of ship in the atmosphere of Mars; 5 - landing ship; 6 - injection of ship into orbit of Mars; 7 - launch to Earth; 8, 9 - rotation of ship for braking; 10 - landing on Earth.
Key: (1). Mars. (2). Earth.

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The angle of the entry of section in the atmosphere of Mars must compose 24° and be supported by constant until the nose section of the section begins to rise upward. From this point on, with the aid of the aerodynamic elevons the angle of attack will be reduced to the value, necessary for maintaining constant flight altitude on the order of 60-80 km.

Upon entry into the atmosphere of Mars thetemperature of the surface of the section in most heated sections will reach, as it is noted in foreign press, 1200°C.

Constant-level flight from 60 to 80 km begins at speed of 3350 m/s and continues for 13 min. Braking in this phase of flight will be provided with a change in the angle of attack until it again achieves 24° (this corresponds to a maximum lift coefficient). Descent in the section will be renewed at a speed of 2400 m/s.

At an altitude of approximately 23 km, when the speed of the section decreases to 450 m/s, the parachute will be opened, owing to which the speed again decreases. The parachute will be discarded at an altitude of 10 km, and the landing engine repeatedly will be switched on.

Landing it is proposed to complete in polar latitudes of Mars, near snow caps, which can be utilized for guaranteeing expedition with water. For the selection of the best landing place it cut off it will have the capability to be moved in the horizontal plane by several hundred meters and to "hover" above the surface for 60 s (Fig. 70).

Landing must be accomplished at a speed of vertical descent of 3 m/s. Energy of landing shock will be absorbed by the destroyed elements, established/installed at the end/lead of the telescopic

landing supports, which ensure the possibility of landing the section on the slopes with slope/transconductance to 15°.

As it is indicated in foreign literature, for duration entire period of reduction/descent crew of section will have capability to discontinue flight, utilizing simultaneously landing and takeoff engines or only takeoff engine.

In the program of experiments of Mars it is planned to conduct biological research on the planet.

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In this case there will be taken measures against the contamination by its terrestrial organisms, carried there by cosmonauts and apparatus, which will make it possible to study the forms of life on Mars in their initial form. Are simultaneously provided for measures against the contamination of cosmonauts and apparatus by harmful Martian organisms.

Investigations of surface of Mars will be first carried out from cabin/compartment. Cosmonauts will observe the visible forms of plant and animal life, then they will remotely take samples of the forms of life and will determine the degree of their harmfulness for the man.

After this, without emerging from section, cosmonauts will determine level of hazard of solar radiation, emission of planet,

composition of gases of atmosphere of planet. After fulfilling the works indicated, cosmonauts will leave to the surface of Mars. For conducting the outlined study program in a radius to visible horizon 20-25 days there will be required three cosmonauts.

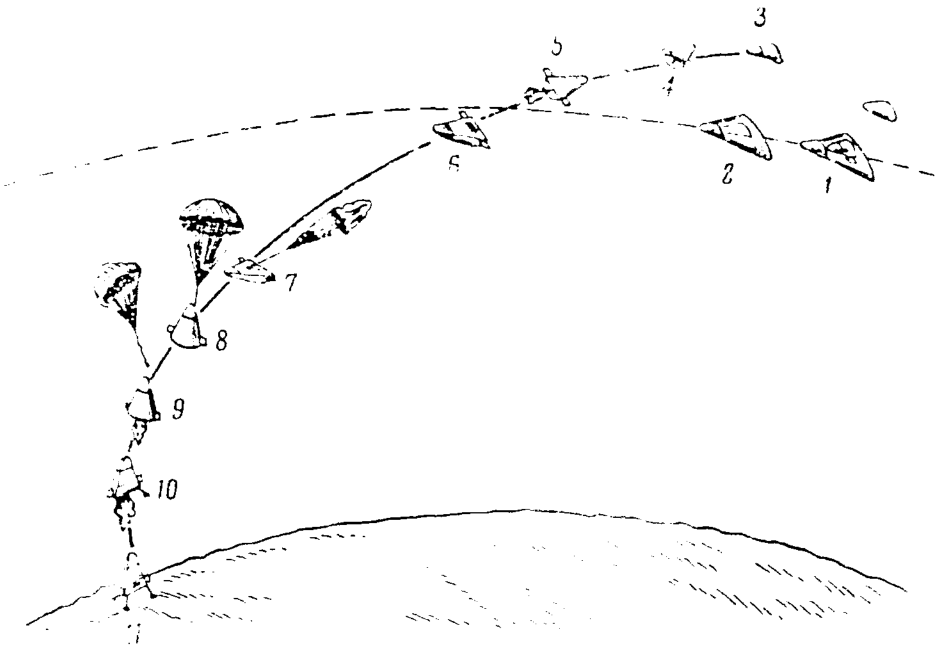

Fig. 70. Sequence of reduction in section with crew to surface of Mars: 1,2 - separation from the ship of landing stage; 3, 4 - entry of landing stage in the atmosphere of Mars: 5 - braking by engines; 6 - braking by aerodynamic forces; 7 - production of brake parachute; 8 - descent by brake parachute; 9 - separation of brake parachute, firing brake motor; 10 - descent power-on; 11 - contact with surface of Mars.

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Cosmonauts must gather data about the structure of the surface of planet, the location of water and volcanoes, take samples of the soil, plant life, etc. All samples must be placed into the reinforced capsules, whose retention is provided for even with the catastrophe of section upon its return to the earth.

a period of stay on Mars the cosmonauts in the takeoff the section, which completed landing, will be built up in and Mars, then they will leave in orbit of space vehicle, s and mating with it will be carried out, they will pass in dump takeoff stage and will start in direction of Earth. ight will be prolonged 240 days.

ing of the section returned to the earth will be made from steel of sandwich construction and outside covered with material with thickness of 9 cm. The internal volume of or positioning/arranging of six cosmonauts and 360 kgf will be approximately 14 m^3.

ion, returned to the earth, for eight hours before landing eparated/liberated from irreversible section and enter into with width of 16 km above dense layers of atmosphere, where "seized" by gravitational field of Earth.

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Chapter 4.

ORBITING SPACE STATIONS.

At present in foreign literature they begin more and it is more than attention to give to orbiting space stations with man aboard for space research. Such stations create favorable conditions for studying the possibilities of person, who is found in outer space, to his ability to carry out observations, to analyze and to fulfill different functions, including not planned prior to the flight. Cosmonaut on it can satisfy operations simultaneously with the automatic devices, backing up and supplementing them. This simplifies the automatic systems for control. The space station and person, who directly participates in its work, can accomplish the most diverse tasks in the space research.

As is known, initially during flights into space primary attention was paid to study of biomedical factors, which appear with prolonged stay of man in space under conditions of weightlessness, and his ability to withstand these conditions. The manned space stations make it possible to expand space and tasks of investigations. For purposes of the successful study of space in the long period on the stations they intend to install the device which simulates gravitational force.

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In the press it is noted that these stations will be used for conducting investigations in the field of astronomy and physics of the sun, since they are not affected by interferences created by the Earth's atmosphere, or for checking equipment and systems of future space vehicles. Briefly stated, the manned space station must become test bench for conducting of different investigations and tests under the conditions of outer space.

For practical purposes it is provided for to utilize space station for observation of cloud cover of Earth (this necessarily for compilation of weather forecasts), conducting of oceanographic investigations, mapping of earth's surface, etc. Space stations can be used for the exploration, the communication, the navigation, etc. Besides this, it is planned to utilize them for the adjustment of promising interplanetary apparatuses, assembly in orbit and the starting/launching of interplanetary flight vehicles.

Is not caused doubt, that space stations can be used for improvement of equipment and systems, util[illegible] for military purposes.

It is noted that operating complex of space station must consist of two fundamental elements: strictly space station with crew and of complex of material and technical supply, into which enter transport space vehicle, carrier rocket and ground stations. Each of these elements consists of many units with different interfering

characteristics. In particular, the orbital parameters have an effect by weight of the concluded load, which is decisive during determining of the dimensions of a space station and quantity of members of its crew. As is known, the altitude and dip angle to the equator are fundamental orbital parameters. Investigations with respect to load change for the two-stage carrier rocket in the dependence on the height/altitude and the dip angle of orbit to the equator show that the peak load can be derived during the starting/launching eastwards, since in this case its own speed of rotation of the Earth is utilized, and the dip angle of orbit is almost equal to launch latitude of rocket.

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If it is necessary to put the carrier rocket into orbit, close to equatorial, then useful load must be decreased due to supplementary fuel/propellant, which is required for rotation of orbit in order to draw it nearer equatorial. During the launching into orbit with a larger dip angle the useful load must be even more reduced (due to the fuel/propellant), since here less will be used its own speed of rotation of the Earth.

During operation of the space station at low altitudes increased expenditure of fuel/propellant for overcoming of aerodynamic drag will be required. However, with an increase in the orbit altitude appears the need for establishing/installing radiation shielding, which entails the decrease of payload weight. Therefore, an important value

is the selection of orbit altitude. The optimum height of orbit can simplify communication, data processing, orbital rendezvous, and rescue operations.

American specialists assume that if space station with crew of 24 people will be found in orbit for five years, then in this case it is expedient every 90 days to replace half of crew members in order to ensure average/mean stay in orbit of each cosmonaut in the course of six months. If the frequency of the replacement of cosmonauts will be caused by the disturbances/breakdowns of the physical state of people as a result of the prolonged effect of weightlessness, it can prove to be necessary to simulate the gravitational force in the period of their determination in the space station in orbit. In connection with this at present in the USA there are conducted investigations of the project of the space station with the creation of artificial gravity (Fig. 71). In this case there will be achieved economic gain because of the fact that the crew members can remain in orbit more prolonged time and, therefore, decreases a quantity of starting/launching of transport apparatuses.

Effectiveness of space station from point of view of cost can be expressed by cost of one man-hour, spent on the conducting of experiments, which can be approximately equal to cost/value of man-day in orbit.

On one crew member in a 24 hour period comes some part of weight

of space station, transport apparatus and all consumables.

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The economic effectiveness of space large-size stations and with the large lifetime in orbit undoubtedly is raised. At the large space stations it is more man- hours, than on the low, it is abstracted/removed for conducting of experiments, at least because each colleague less than the time expends on domestic service.

They consider that during operation of space station to 44% of expenditures it falls for technico-material supply; large part of these expenditures/consumptions is caused by high cost/value of carrier rockets of one-shot application. The use of carrier rockets of repeated application, obviously, decreases the expenditures/consumptions, connected with the material and technical supply, and will allow more bending to accomplish/realize operation the space manned station.

Is examined possibility to utilize as transport apparatuses for delivery/procurement to space station of people and loads apparatuses of type "Gemini" and "Apollo". There is also conducted a study of a 12-local transport apparatus, which develops lift upon atmospheric entry.

Fig. 72 shows an orbital station with a crew of 24 cosmonauts.

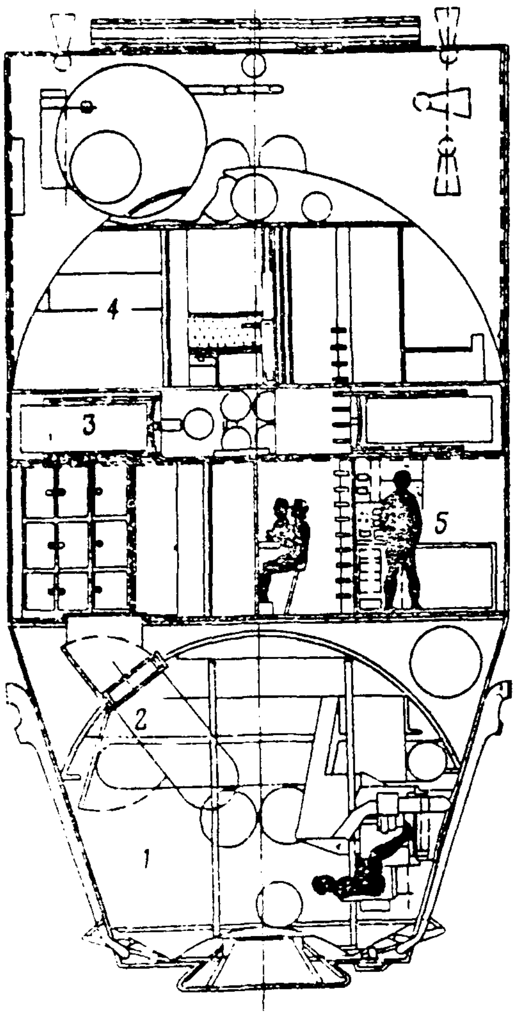

Fig. 71. Design of a six-place space station with a centrifuge for the simulation of artificial gravitational force: 1 - hangar for reception of transport apparatus; 2 - air lock; 3 - centrifuge; 4 - everyday locations of crew; 5 - laboratory for conducting experiments.

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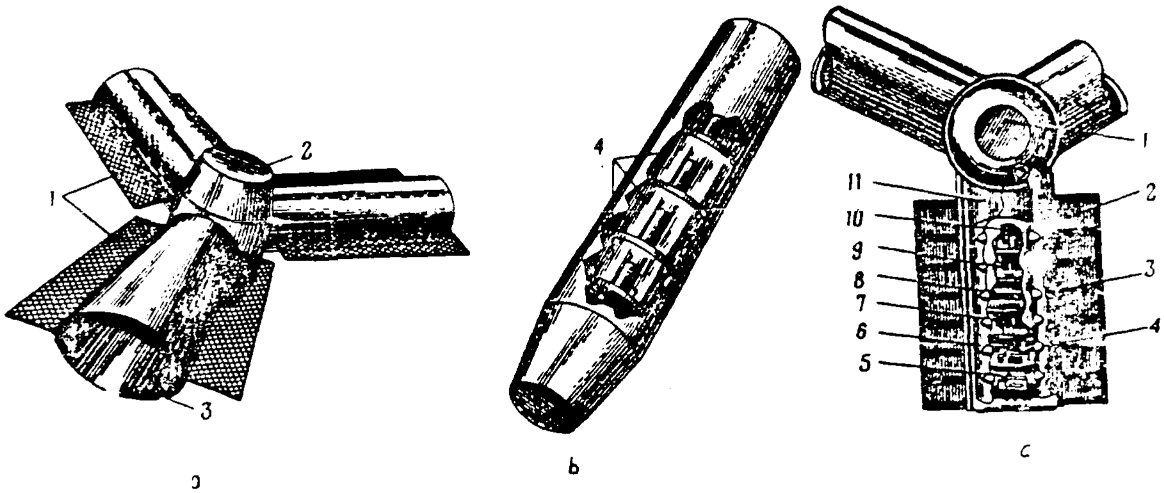

Fig. 72. Orbital station with crew of 24 cosmonauts: a) general view; b) cross section of nonrotating part; 1 - solar blade; 2 - central nonrotating cylinder; 3 - cylinder ("blade"); 4 - compartment in central nonrotating part; c) cross section of "blade": 1 - nonrotating part; 2 - solar blade; 3 - duct; 4 - air lock; 5 - habitable compartment; 6 - command compartment ; 7 - laboratory compartment; 8 and 9 - working compartments; 10 - storage; 11 - shell, which shields from radiation and meteoritic particles.

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According to its general view, it is similar to a propeller. Station consists of three radially located cylinders ("blades"), fastened to the central cylinder. Radial "blades" rotate at a rate of 4 r/min, which provides the creation of g-force 0.4 g in the most extreme

compartments of the "blades". Each "blade" consists of six compartments with a diameter of 4.5 m each. In the central cylinder the laboratory for conducting the experiments under the conditions of weightlessness is placed. In its upper part hermetically sealed location for the maintenance of transport apparatus is located.

Fundamental problems during design of space stations is providing safety of man and his return to the earth, efficiencies of man under conditions of weightlessness during prolonged period, and also normal work life-support system. Due to the weight limitations of the space stations, concluded in orbits, it is necessary to use regenerative system for obtaining of oxygen and water. As it is indicated in foreign literature, in the first stations carbon dioxide in the compartment is intended to be removed by adsorption, and, subsequently, to use the chemical methods, which ensure the obtaining of carbon and oxygen in the final analysis. The transformation of the dioxide of carbon and hydrogen in the presence of catalyst into carbon and water is provided for by one of such methods. During the subsequent electrolysis from the water will be obtained hydrogen, which then they will utilize in this process, and oxygen, which will enter flight deck. The process of obtaining drinking water from contaminated water and urine is developed. It is reported that already satisfactory results are obtained.

As sources of power supply chemical batteries, fuel cells and solar batteries are provided. The selection of the source of power

supply will be determined by the duration of its functioning, by the value of load and by duration of flight. In particular, for the short-term flights with the moderate consumption of electric power it is possible to utilize fuel cells.

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For the more prolonged flights with the high energy consumption, apparently, it is expedient to allow the combined use of solar and chemical batteries.

System of position control of space station provides for use of managers of nozzles for rough orientation and inertial flywheels for precise orientation.

Programs of works on the orbital stations.

In the USA there are several programs of works on orbital stations. On to the so-called program MOL (system of orbiting space station) is provided for the investigation of the possibility of designing of orbital station with the personnel of 2-4 people and his monthly replacement (Fig. 73).

Station has cylindrical form, its length about 5 m. In orbit the station will be derived/concluded without the cosmonauts. For the delivery/procurement to it of cosmonauts and equipment, necessary with the work, they intend to utilize the spacecraft, which consists of the

section, in which is placed the crew, and cargo hold. For replacing the cosmonauts it is planned to utilize the transport spacecraft, concluded in orbit by the existing rockets.

According to another study program it is proposed to create station with personnel of 4 people, but designed for year of work. This station will have cylindrical form, the length of its 6.9 m, weight - 7-9 T.

Fig. 73. Orbital station with a crew of two people (according to the program MOL).

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Station will be placed in a circular orbit with an altitude of 320 km without the cosmonauts. For the delivery to it of cosmonauts, it is planned to utilize "Gemini" satellites, while for the delivery of loads - special pilotless apparatuses with a weight of up to 2.3 t.

There is also a program, which foresees the investigation of the possibility of using the "Apollo" spacecraft as an orbital station. Personnel of this station must consist of three people, the duration of its operation is 100 days. In this case are examined two versions of the arrangement/position of the cosmonauts: in the crew compartment and in the modified power bay.

There are still several programs which investigate possibilities of designing of stations, designed for 1-5 years of work with personnel into 24-30 or 18-36 people. The station, whose crew

consists of 18-36 cosmonauts, will weigh 96 t. In appearance it will be similar to a three-bladed propeller. The "blades" will be made removable, and the spread of them is 45 m. Prior to injection into orbit they will be forced against the center section, and they are removed in orbit. For the creation of artificial gravitational force to "blades" will be given rotation at a rate of 4 r/min. Each "blade" will be divided into six sections with a length of 2.4 m each. It is assumed that the station will be injected into orbit by the height/altitude of 300-500 km. The supply of station and the delivery/procurement to it of cosmonauts will be accomplished/realized by the spacecraft, started in 90 days or somewhat more frequent. It is noted that for the supply of station can be used the "Apollo" spacecraft. Of this case the ship must consist of the crew compartment, in which can be placed six cosmonauts, and cargo hold. The first section is created on the base of crew compartment, the second - on the base of the power bay of this ship.

In crew compartment cosmonauts will be placed in two rows of three people each, and they will be breathe pure oxygen.

Upon entry of the crew compartment into the atmosphere and with descent to the earth by its apparatus at a specific angle of attack, it is possible to achieve a lift-drag ratio equal to 0.5. Utilizing the orientation system and lift of section, the cosmonauts can ensure its landing at any point in the band with a length of 2380 km and with a width of 560 km.

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In the cargo hold, besides loads, will be placed main engine of ship and orientation system, which ensures approaches and rendezvous of ship with orbital station, and also certain equipment life-support system. The actuating elements of the orientation system are four ZhRD with a thrust of 45 kgf each. Each engine has four nozzles, located in cruciform. The main engine and ZhRD of orientation system must ensure the relative speed of ship and orbital station at the moment of mating not more than 0.6 m/s.

After spacecraft docking with station, which cosmonauts can observe visually through window in cover of hatch of spacecraft, this cover will be opend and cosmonauts will have capability to pass into location of station and to transpose there loads.

Ship without crew will for a certain period of time (to 6 months) remain that moored to station. It must be ready to the return to the earth to any moment.

Flights of manned space station of this type abroad consider as a means of evaluation of potential possibility to accomplish in outer space actions of a military character. In this station it is decided to maximally utilize finished articles, aggregates/units and systems of other spacecraft, in particular, the finished elements of spacecraft "Gemini" and by "Apollo" in order to reduce the cost of the station to a minimum and the risk of the unchecked solutions during

the development.

An important advantage of the station, provided for by program MOL, is considerably increased free volume for crew activity. This, by hypothesis of the authors of project, in the first place, it will make possible to crew to accomplish/realize greater physical activity than on the ships "Gemini" and "Apollo"; secondly, it will contribute to the maintenance of a good health of crew members in the prolonged space flight; thirdly, it will make it possible to fulfill different combined experiments.

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The station is intended to utilize for the tests and the evaluation of the experimental models of space systems and components under the actual conditions for orbital flight, which will make it possible to more rapidly and more economically develop/process more reliable equipment.

Fundamental experiments planned by program MOL are directly connected with promising application of space technology and are intended for determining role of man under conditions of outer space.

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Chapter 5.

PRESSURE SUITS FOR THE COSMONAUTS.

For first spacecraft with man aboard in the USA, there were created pressure suits, which provided to cosmonaut the possibility of continuous flight during 24 hour. The fundamental elements of this pressure suit are: housing, space helmet, gloves, boots and linen next to the skin. Weight of pressure suit is 9 kgf. Its body consists of two layers: internal (hermetically sealed) from the nylon fabric with the special coating and external (power) from the same fabric with the aluminized coating.

For putting on of pressure suit in its body is made hermetically closing with the aid of fastener "lightning" cut along a diagonal, which goes from left arm to right thigh.

Ventilation system of pressure suit is made thus. On the internal layer of the body there are attached tubes prepared from the wound spiral wire, covered from above by rubberized nylon tape. In them there are openings with a constant step/pitch. Gas enters the pressure suit through the branch located on the front above the waist. From the pressure suit the gas is brought out through the output, placed in the occipital part of the space helmet.

Space helmet is made of glass-fiber-reinforced plastic. It has a swivel feature on a neck ring, sealed rigidly in the body and ensuring fastening of the space helmet. Gloves are fastened to the hoses/pipes of pressure suit and are fixed/recorded with ball bearing locks.

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The miniature electric bulbs of red color for the dashboard light and maps in flight in the shadow of the Earth are assembled at the ends of the indicating and middle fingers of both gloves. The light boots are prepared from nylon fabric with aluminized coating and rubber feet.

In the case of emergency depressurization of the cabin/compartment of the ship in pressure suit there must be maintained an excess pressure of 181 mm Hg.

In this pressure suit American astronauts completed flights aboard spacecraft of type "Mercury" with duration to 34 hour.

For flight service to longer duration, and then for the putting of a cosmonaut into open space, it was necess to create pressure suit, which simultaneously had to be emergency covery facilities of cosmonaut (during depressurization of ship) and means of autonomous existence of man in outer space. Was first proposed the pressure suit, which consists of three layers: an inner (hermetically sealed) layer from the rubberized nylon, a middle (power) layer from a nylon grid and an outer layer, also from the nylon. The weight of pressure

suit was 11.9 kgf. In such pressure suits completed flights the cosmonauts aboard the ship "Gemini".

Subsequently into this pressure suit were introduced changes for purposes of facilitation of its putting on, especially space helmet (under normal conditions for flight cosmonaut was not put on space helmet). For guaranteeing the output into space there were added two additional layers, which ensure thermal insulation and meteoritic protection, and on the space helmet there was installed a supplementary shield and lowered deflector for the protection from the ultraviolet radiation of the sun. Space helmet is made from organic glass. Weight of this pressure suit 14 kgf.

Thermal insulation of pressure suit consists of seven layers of polyester aluminized film with appropriate separators. As it is indicated in the foreign press, the film reflects the greater part of radiant solar heat, and separators block the contact between these layers during the heat transfer via thermal conductivity. Meteoritic protection is provided by one layer of felt. This layer is placed between the power and outer layer of pressure suit. The sight glass of space helmet is the most vulnerable place of pressure suit from the point of view of thermal insulation.

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For protection from meteoritic particles on the glass of space helmet outside there is installed a shield from the very durable material -

Lexan. On the shield a sealed coating, which gives yellowish coloration to glass, will be applied.

On going into space the cosmonaut hermetically sealed gloves are put on top of heatproof gloves, since, according to the information of foreign press, the temperature of sunlit surface of the ship was equal to +128°C, and on the shadow side of ship it was equal to -168°C.

In outer space the cosmonaut emerged from the ship "Gemini" in a pressure suit with a breast pack of life-support system. Oxygen from onboard of ship on a pipeline with a diameter of 6.3 mm, laid within the attachable halyard, entered into the pack. In the case of the cessation/discontinuation of the supply of oxygen on the conduit/manifold in the haversack, there was an emergency oxygen supply system, which ensures the respiration of cosmonaut during 10 min. This is completely sufficient time so that the cosmonaut could return aboard the ship. The dimensions of the pack are 33x15x5 cm and weighs 3.6 kg.

Oxygen, passing through the pressure suit, provides heat removal in quantity of 175 kcal/h and the removal from it of carbon dioxide and moisture. Oxygen from the pressure suit is sent into outer space.

In the foreign press it is reported that all subsequent egresses of cosmonauts into outer space were produced with packs which have stored up oxygen of 30 min.

For a 14-day flight of spacecraft "Gemini" there was developed a simplified version of the pressure suit, which has two layers - hermetically sealed from nylon and power from high-temperature (strength) nylon fabric.

Pressure suit is equipped with air channels for distributing oxygen flow throughout entire body, including finiteness. Instead of the removable rotary space helmet to it is fastened on the fastener "lightning" the soft canopy from the fabric, into which is sealed the sight glass. Before putting on the hermetically seal pressure suit, the cosmonaut must put on the head the shielding helmet of the pilot. Soft boots with rubber feet are put on the struts above the pressure suit, and hermetically sealed gloves are fastened to hoses with the aid of the circular hinge. All of this pressure suit, including the weight of helmet, 7.2 kgf.

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Each crew member of "Apollo" spacecraft it is proposed to supply with pressure suit, which must serve as emergency means during 115 hour (in the case of service failure life-support system of ship or partial destruction of a pressurized cabin of the crew) and ensure the autonomous existence of cosmonaut during 4 hour after his egress to the surface of Moon. Since for cosmonaut on the Moon it is necessary to satisfy the specific actions (independently to be lowered from the lunar section on the u-bolt to the surface of the Moon, to walk over

its surface, to assemble the lunar samples, etc.), then the construction/design of pressure suit must provide the necessary mobility in the presence in it of excess pressure. The design of the pressure suit must be, furthermore, adapted to temperature conditions existing on the Moon. On the Moon there is no atmosphere, its surface undergoes wide fluctuations in temperature: from +100-150°C in the daytime to -100-150°C at night, to the effect of cosmic rays, meteorites, X-ray and ultraviolet rays, emitted by the sun. The gravitational force on the Moon comprises the sixth part of gravitational force on the Earth.

At first the pressure suit was developed with the system of gas cooling, but it did not provide cooling body of cosmonaut on calculated levels of physical activity on Moon (400 kcal/h under normal conditions and 500 kcal/h with peak loads). It was subsequently decided to use the closed system of water cooling.

Lunar pressure suit of "Apollo" spacecraft (Fig. 74) has six layers:

1) linen with water cooling;

2) backing/block from very flat nylon;

3) grid with attached to it air ducts for ventilation of housing, hands and legs;

4) hermetically sealed layer from dense nylon with special coating, which has rubber spirals in places of coupling;

5) outer layer from light nylon, colored light blue;

6) layer of heat insulation and protection from micrometeorites, made in the form of jacket with hood also of trousers.

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Weight of pressure suit is 13.6 kg.

Space helmet of pressure suit provides free rotation of head of cosmonaut in any direction. For defogging it is ventilated by air. Before putting on the pressure suit, the cosmonaut must put on a light leather helmet with earphones and a microphone to the head. He puts on the pressure suit in 5 min through the dual cut on the spin with the fastener "lightning".

On leaving the surface of the Moon the cosmonaut puts on a jacket and trousers, which shield him from solar radiation and flows of micrometeorite particles, lunar boots, heatproof gloves, spherical space helmet (above flight interphone headset) and portable life-support system. Shielding jacket and trousers have outside a layer from white capron, which reflects solar rays, seven layers of thermal insulation and two layers of nylon with a special coating. The insulation is reinforced on the elbows. Heatproof gloves in the region of the palms consist of sixteen layers of thermal insulation and above them a layer of fabric from the noncorrosive steel filament for an increase in strength of gloves by the wear. The foot of lunar boots is made from silicon rubber and thirteen layers of insulation/isolation from felt. During the tests the boots provided

heat shielding at a temperature of 120° and protection of struts during walking over the uneven surface with the cutting edges during more than 1 hour.

Fig. 74. Lunar pressure suit: 1 - coupler; 2 - power shell; 3 - sealing layer; 4 - linen with system of water cooling; 5 - heatproof layer; 6, 8 - beta-filament; 7 - special film; 9 - boots.

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Space helmet is prepared from polycarbonate and is covered outside with gold film for protection from misting and reflection of ultraviolet rays, emitted by sun. From the front on the housing of pressure suit there are two branches. To one branch is connected the hose of the system of gas ventilation, to another - the hose of water cooling system. Furthermore, is a connector plug for the connection to the onboard communication system.

The haversack of life-support system has following dimensions: length of 45 cm, height/altitude of 66 cm, thickness of 26 cm. Its weight is equal to 29.5 kgf. The jacket/housing of the portable system is made from glass-fiber-reinforced plastic. Its internal wall is shaped on the back of the cosmonaut: the external wall is given a cupped form so that the cosmonaut could be rolled over the lunar surface, if he falls to the back.

System of water cooling consists of pressurization small tank, centrifugal pump with electric motor of direct current, storage accumulator of water, sublimator (for cooling of water and air) and linen water-cooled. Supply system with oxygen maintains in the pressure suit excess pressure, while the system of the circulation of oxygen on the closed circuit provides removal from the atmosphere of the pressure suit of carbon dioxide, moisture and odors.

Compressed oxygen under pressure of 63 kgf/cm^2 in a quantity of 0.5 kgf is stored in a cylindrical tank made of stainless steel.

For the circulation of oxygen with a constant speed, a fan, which creates a pressure of 7.5 mm Hg, is used. Moisture is removed from the gas flow in the separator and drains from it into the water small tank. Cylindrical container for absorbing of carbon dioxide and odors contains hydroxide of lithium and activated carbon.

A tank with emergency reserve of 91 gf of oxygen at a pressure of 616 kgf/cm^2 is stored in the portable life-support system. This quantity, as they assume/set, it is completely sufficient in order to return into the lunar it cut off during the considerable removal/distance from it. Oxygen is ducted into the space helmet. Emergency oxygen system can supply oxygen, also, into the primary system of the supply with oxygen.

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In the upper part of the haversack system of two-way vocal radio communication with cosmonaut, who remained in lunar section, is assembled.

Duration of continuous work of portable life- support system is 4 hours. Repeatedly servicing oxygen cylinder and water system, and also replacing container with lithium hydroxide, the cosmonauts can make several egresses to the surface of Moon with one and the same portable life- support system.

In the foreign press it is indicated that for the pressure suit of "Apollo" spacecraft there was proposed a special harness, which makes it possible for the cosmonaut to endure 30--fold g-forces in any direction. Cosmonaut in this case manages without a seat. With the aid of the harness the pressure suit is fastened to the guides, on which is accomplished/realized the movement of cosmonaut. From the front g-force is received by cloth skin/sheathing of pressure suit,

and from behind also along the sides - by aluminum honeycomb design between layers of glass-fiber-reinforced plastic. They assume that the pressure suit of this construction/design will find use aboard the spacecraft upon hypersonic entry into the atmosphere or during supersonic low-altitude flights under conditions of high turbulence.

It should be noted that pressure suits, prepared from fabrics in combination with rubber layers or couplings, have shortcoming. With in them in the excess pressure increase of gaseous medium they are inflated and impede the motion of cosmonaut. Furthermore, with the prolonged carrying they are rubbed on the elbows and the elbows; are not excluded also punctures and cuts of a hermetically sealed layer. Therefore are at present developed/processed also rigid space pressure suits. In one of these pressure suits the glass-fiber-reinforced plastics with the honeycomb fillers and aluminum joint rings are used.

Future interplanetary spacecraft can satisfy flights with a duration of one to three years with a crew of 6-14 people. In this case the flight decks will have the artificial atmosphere of 50% of oxygen and 50% of nitrogen at certain excess pressure. However, the crew will complete the flight without the pressure suits. But for different operations outside the ship pressure suits of different types now are developed/processed.

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Proposed were pressure suits both of soft construction and hard, with

portable life-support systems. In the process of studying the soft and rigid pressure suits appeared the shortcomings and each type advantages. At present in the USA there is created a pressure suit of composite construction: its upper part rigid, lower - elastic. Life-support system will be placed in the rigid part of the pressure suit, which, as they assume, will considerably improve the maneuverability of the cosmonaut in open space. The pressure suit will provide a stay in outer space for 4 hours.

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Chapter 6.

FUTURE MANNED HYPERSONIC FLIGHT VEHICLES.

Together with the intense development of aircraft n last decade they were created and carrier rockets and space vehicles received wide distribution.

The launching of space vehicles in orbit of artificial Earth satellites and to other planets made it possible to achieve flight speeds of 28000-40000 km/h, altit.des of 300-1500 km, and distance from Earth - up to hundreds of thousands and millions of kilometers.

The question involuntarily arises: why the aircraft, whose development laid the groundwork for the creation and launching of artificial satellites and spacecraft, considerably fell behind in speed and flight altitude of the newly originated flight vehicles. The fact is that the rocket engines make it possible to obtain enormous powers for the very short operating time and to accelerate apparatus to the requiring speed, which ensures its flight in outer space, where is absent aerodynamic drag and, therefore, it does not occur heating apparatus due to braking of atmospheric air. The passage of the atmosphere on leaving in orbit and descent from it is accomplished/realized for the small time interval. Materials, from which are prepared the engines, and corresponding protection of

apparatuses provide with them reliable efficiency under these severe conditions.

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However, realization of the endurance flight for very high speeds (with number M=7-8) proved to be more difficult task because existing materials cannot withstand thermal loads appearing in this case.

As a result a break between speeds of 3000-3200 km/h, at which there is completed an endurance flight of contemporary aircraft, and velocities of endurance flight of satellites in low circular orbits (approximately 28000 km/h), was created, and also a break between flight the altitude of aircraft (approximately 25 km) and flight altitude of space vehicles - 180-200 km.

It was natural to expect that these speed ranges and heights/altitudes cannot remain completely unmastered, but they will be conquered. And, actually, in the foreign press it is noted that the intense mastery of these speeds within the next few years will be begun. It already occurs in two directions. The first - this is an increase in speed and flight altitude of aircraft during their prolonged sustained flight at the hypersonic speeds. The second - development of the so- called aerospace vehicles, which from the orbit descend in the atmosphere and complete in it partially inertial, partially aircraft flight. In other words, one direction is occupied by the apparatuses whose speed will grow from the relatively low

values to the hypersonic over the long term, and the other - by apparatuses whose speed with the orbital will be reduced to the hypersonic, and which then will make a landing as a standard aircraft.

However, what does this mean for flight vehicles? What are their possible types, special features of layout and design?

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On the specific special features of flight these apparatuses can be divided into the following three classes:

1) the march hypersonic aircraft, on which is feasible prolonged powered flight in free-running speeds and at altitude;

2) the hypersonic booster aircraft, not fitted out to the endurance flights at the hypersonic speeds; the fundamental flight conditions of these aircraft - continuous dispersal/acceleration due to the work of engines to the prescribed/assigned limiting value of speed;

3) the aerospace vehicles, concluded to the orbital speeds either close to them with the aid of the carrier rockets or the booster aircraft; after uncoupling with the aerospace vehicles the booster aircraft enter or the dense layers of the atmosphere, they complete gliding flight, maneuvering due to the aerodynamic forces and the thrust forces of engines and are set on the prescribed/assigned airfield; the characteristic feature of their flight - continuous braking in, as a rule, with inoperative engines.

In principle, judging according to the press, there can be a fourth class of aircraft, which unites the last two. The flight vehicles of this class themselves are accelerated/dispersed to the orbital speed, complete flight into space, they converge from the orbit into the dense layers of the atmosphere, they maneuver, are braked and are set as usual aircraft.

We will be restricted to examination of hypersonic aircraft of first three classes.

CRUISING HYPERSONIC AIRCRAFT.

They consider that cruising hypersonic aircraft can find use in transport and military aviation. Their cruising speed can reach number M=15 at the altitude of 60 km.

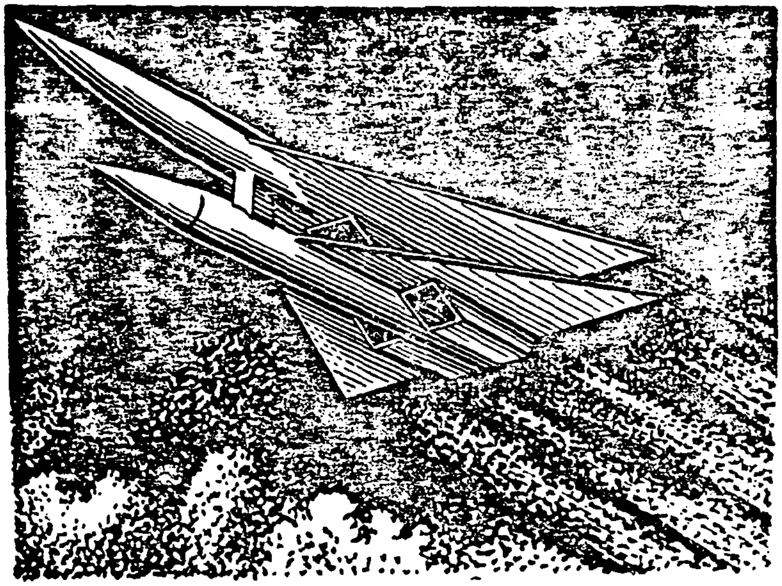

Fig. 75. Cruising composite hypersonic aircraft designed for a flight speed which corresponds to number M=7.

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Fig. 75 shows exterior view of cruising composite/compound hypersonic aircraft, designed for flight speed, which corresponds to number M=7.

Project of one of hypersonic aircraft, designed for flight speed, which corresponds to number M=15, and altitude of 60 km, provides for installation of turbojet and ramjet engines. Takeoff is accomplished with a turbojet engine. With number M=3 there is switched on the ramjet engine, which accelerates/disperses aircraft to number M=15. The combustion chamber of ramjet engine surrounds the fuselage of aircraft.

Depending on flight speed weight balance of cruising hypersonic aircraft can substantially change. Fig. 76 gives the weight distribution of hypersonic transport aircraft with the flying range on the order of 5000 km in the dependence on the Mach number. The payload of such aircraft, judging according to given in the press data, with an increase in the rated speed noticeably decreases and with number M=7 composes only 2-3%. The structural weight insignificantly grows/rises. However, the weight of power plant virtually little depends on the rated speed of flight. The weight of the total fuel reserve grows/rises from 45 to 49% with an increase in the rated speed of flight from number M=2 to M=7. Moreover the available fuel reserve to the cruising (cruising) flight with an increase in the rated speed noticeably decreases due to a considerable increase in the expenditure of fuel/propellant for the climb and acceleration. Consequently, for an improvement in the flight characteristics of this aircraft it is necessary to raise the efficiency of the engines in the process of dispersal/acceleration and under cruising conditions.

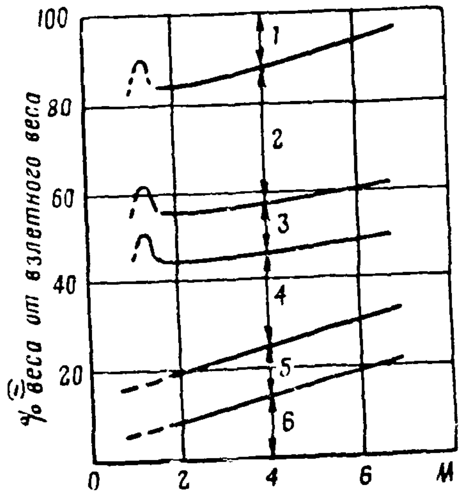

Fig. 76. Weight distribution in cruising hypersonic aircraft depending on Mach number: 1 - payload; 2 - structural weight; 3 - weight of engine; 4 - fuel load; 5 - fuel/propellant to landing and standby fuel/propellant; 6 - fuel/propellant to climb.

Key: (1). % of weight from the takeoff weight.

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A noticeable effect on the economy of flight of hypersonic cruising aircraft is exerted by the acceleration during climb. Obviously, the less the acceleration, the prolonged the process of the climb, and this means, the greater the fuel consumption in this case.

After increasing acceleration, it is possible to considerably decrease fuel consumption. However, this is correct to the specific values of accelerations. The fact is that with an increase in the acceleration the expenditure of fuel/propellant for the climb decreases, but increases the weight of power plant. The latter is

caused by an increase in the engine thrust for obtaining higher acceleration. As a result it is found that the sum of the weight of power plant and expendable fuel/propellant for the climb and cruise with an increase in the acceleration first decreases, and then increases.

HYPERSONIC BOOSTER AIRCRAFT.

Basic goal of study and developments of booster aircraft is the creation of repeated multipurpose carriers for injection into orbit of different loads. It is assumed that these aircraft must after takeoff from the usual airfields accelerate/disperse the subsequent stage or steps/stages to the speed, which corresponds the minimum to number M=7-8, then to return as usual aircraft, to the airfield.

But what are the most important advantages of booster aircraft over carrier rockets, which accomplish virtually the same tasks? Let us examine for the answer to this the fundamental question how there changes the fuel consumption in the process of boosting the stage (stages) to the orbital speed with the rocket and aircraft launches.

It proves to be that dispersal/acceleration to one and the same speed by these two methods requires different quantity of fuel/propellant. For example, the dispersal/acceleration of three-stage carrier rocket to a speed equal to 30% of the orbital requires fuel consumption which is 50% of launching weight of rocket

system, and to the orbital speed - 84% of launching weight. Dispersal/acceleration with the aid of the carrier aircraft to 30% of orbital speed requires the expenditure/consumption, whose weight corresponds only 7% of launching weight, and to the orbital speed - 65% of launching weight.

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As we see, the difference in the fuel consumption very essential.

Advantages of aircraft start in comparison with rocket indicated are caused mainly by efficiency of engines used. With the rocket start utilize the liquid- propellant rocket engines; whose efficiency is relatively low. Even application of such highly efficient fuels/propellants as liquid hydrogen and liquid oxygen, makes it possible to obtain the specific impulse of order 420-450 s (depending on the schematic of engines). On the booster aircraft to number M=14 can be used the air-breathing or compound engines: direct-flow turbine, turborocket, direct-flow rocket, whose pulse is within the limits of 1500-2500 s, but it can reach value of 4000 and even 5000 s in the velocity band M=2.5.

It is terminated, weight of jet engines is more than weight of rocket engines. In spite of this, the value of the payload, put into orbit with an aircraft launch, is higher than that with the rocket. According to data of R. Lane, the payload weight, put into orbit by a carrier rocket, is approximately 5.8% of launching weight, and by a

booster aircraft - 9.5-10.5%. In this case as the fuel both in the first and the secondly the cases is utilized liquid hydrogen.

Noticeable effect on value of concluded payload in orbit with aircraft start exerts speed at the end of starting section, in which worked jet engines. Foreign specialists indicate the advisability of using jet engines to number M=8-9, since with an increase in the velocity at the end of the acceleration phase of booster aircraft the concluded payload grows/rises.

Foreign aviation specialists, mainly American, proposed several designs of booster aircraft.

Fig. 77 shows a diagram of a booster aircraft of one of projects. Aircraft is the repeated carrier of the rocket stage of one-shot use and spacecraft with the lifting body. Entire system works as follows.

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The aircraft is accelerated with engines installed on it to a definite velocity and is disengaged from the rocket stage. After this, the booster aircraft returns to the base. Rocket stage is accelerated to the orbital speed, at which from it is separated the spacecraft.

The design shown in Fig. 78 is very interesting. Aircraft consists of two booster stages and spacecraft, intended for the injection into orbit. All three objects - two stages and a spaceship

- are piloted. The aircraft can be made in the form of cluster (view from left) and parallel connection (view from right).

First aircraft is accelerated with the first stage. Its uncoupling with the second stage occurs at a uniform velocity. First stage returns to the airfield. Aircraft is accelerated with the second stage to the orbital speed. Spacecraft is separated from the aircraft. Aircraft, maneuvering in the atmosphere, returns to the base.

In one of American aviation journals, as the booster aircraft the apparatus shown in Fig. 79 was proposed.

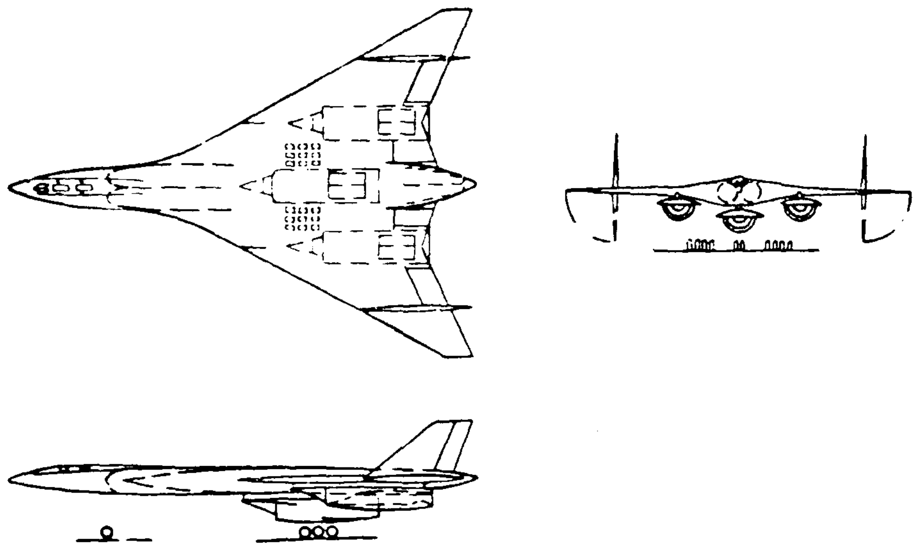

Fig. 77. Diagram of a booster aircraft, which carries a rocket stage and spacecraft.

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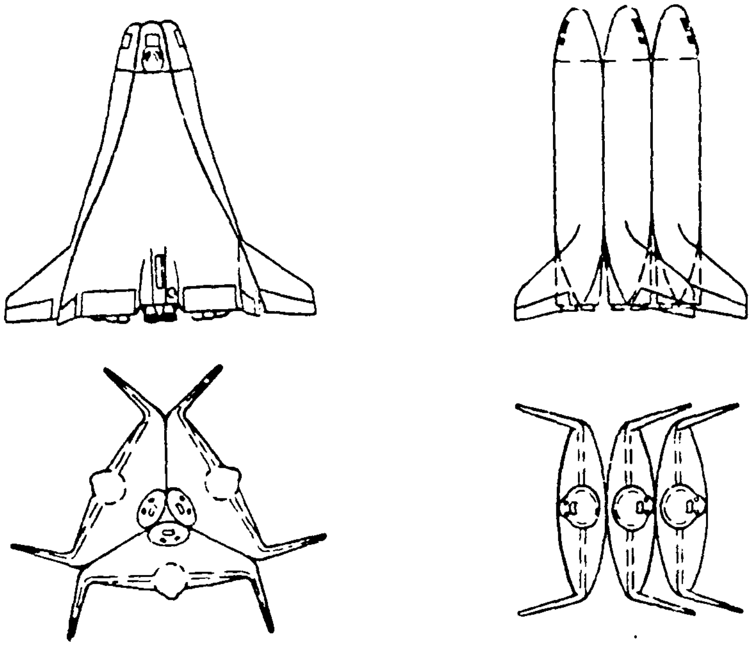

Fig. 78. Booster aircraft which consists of two booster stages and a spacecraft.

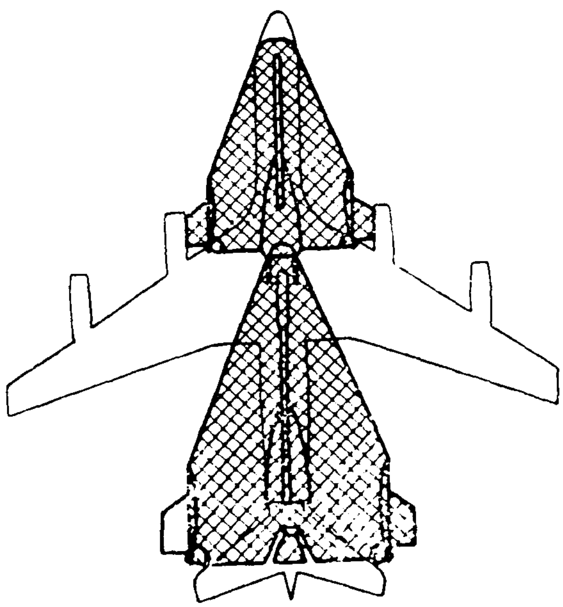

Fig. 79. Booster aircraft with consecutive stages.

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This booster aircraft is two consecutive stages, which have the form of a delta wing. Lower (first) stage first accelerates the aircraft, then it is separated and returns to the place of start. Upper (second) stage is simultaneously a space vehicle. Its power plant provides the output of apparatus in orbit, the accomplishment of maneuver for it, reduction/descent from the orbit and return to the earth.

Another proposed aircraft with second stage, which is also space vehicle, is shown in Fig. 80. The second (upper) stage bears 10 passengers and loads, intended for the delivery/procurement to the space station. Below in the figure booster aircraft is shown at the moment of start, above - at the moment of stage separation.

After uncoupling booster aircraft returns on to bass.

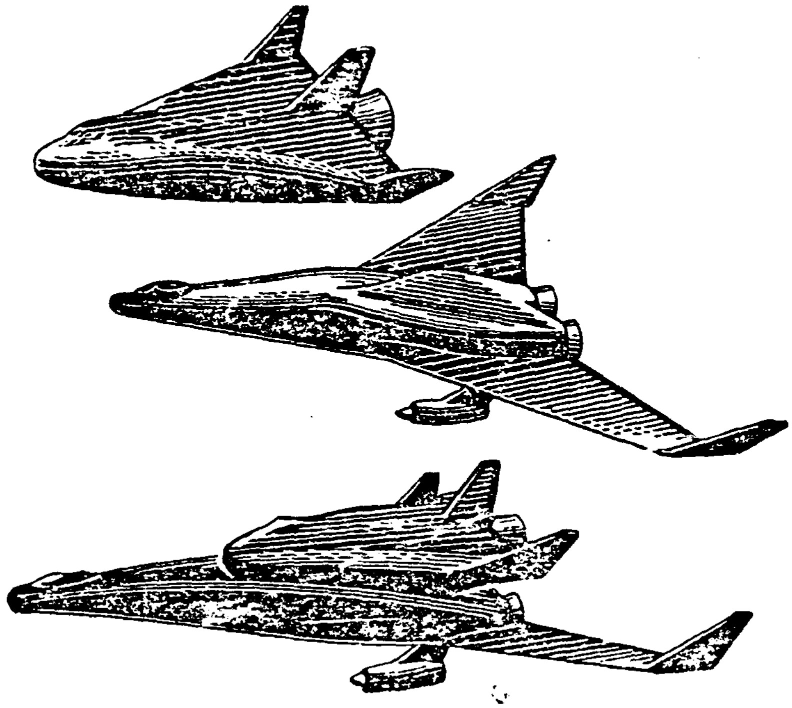

Fig. 80. Booster aircraft with second stage, which is space vehicle..

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The second stage with the aid of the engine, established/installed on it, emerges in calculated orbit and after accomplishing of target returns to the earth under the conditions of gliding/planning power-off. Its lift-drag ratio composes approximately 2.

For delivery of the load in low orbits there is proposed a design of apparatus for repeated application, whose general view is given AN Fig. 81. Figure shows the overall dimensions of aircraft.

Aircraft consists of two steps/stages. First stage has a length

of 53 m, the second - 48 m. The diameter of the fuselage of each stage is equal to 9 m. Sweep angle of first stage of 60°, the second - 65°. First stage accelerates/disperses aircraft to the definite velocity, is disconnected with the second stage and returns to its airfield. The second stage is accelerated/dispersed to the orbital speed, and then returns to the base.

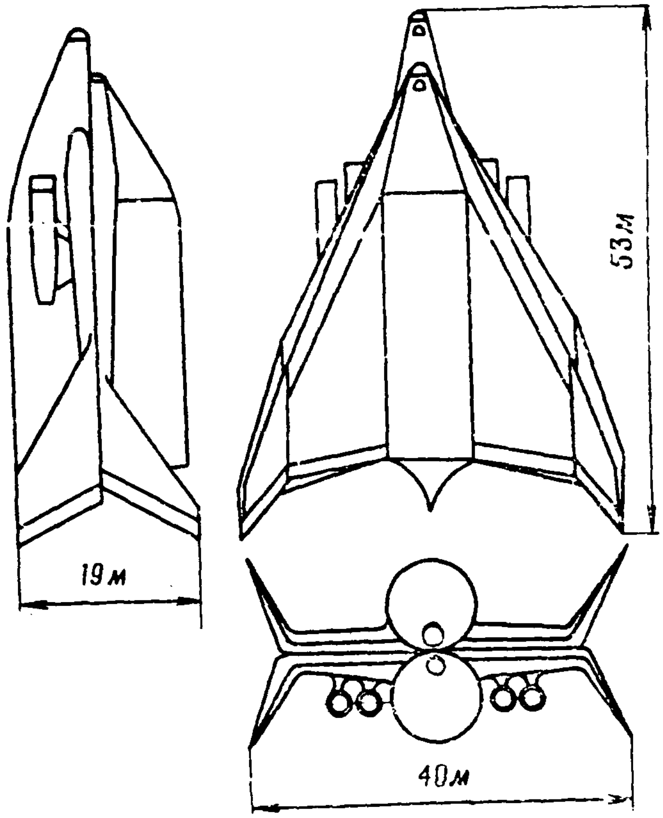

Fig. 81. General view of booster aircraft intended for flights in low geocentric orbits.

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Fig. 82 gives a general view of a booster aircraft of repeated application on project of the firm "Lockheed". Booster aircraft is two winged stages located in parallel. It is intended for the delivery into orbit of 500 km of ten cosmonauts and 3 T of load. The power plant of the first stage consists of a VRD and rocket engines. As the fuel/propellant for the VRD there is utilized kerosene, for the rocket engines - liquid hydrogen (combustible) and liquid oxygen (oxidizer).

After being separated from second stage, first stage returns to the earth with the aid of TRD installed on the wing. The second stage is accelerated to the orbital speed, in orbit rendezvous with the space station and will be joined with it. After the replacement of cosmonauts and unloading of load it returns to the earth under the conditions of gliding/planning.

The American firm "Martin" proposed design of booster aircraft of repeated application for removal of loads in low geocentric orbit. Its general view is shown in Fig. 83. Aircraft has two stages. It is launched vertically. First stage after uncoupling returns to the base, the second stage is accelerated to orbital speed and supplies load to the point of destination, after which it returns to the earth.

Fig. 84 gives a diagram of the two-stage booster aircraft intended for delivery of the cosmonauts and loads to space stations, which are found in orbit by altitude of 300 km. Launching weight of aircraft is 100-200 T. Aircraft consists of two winged stages located in parallel.

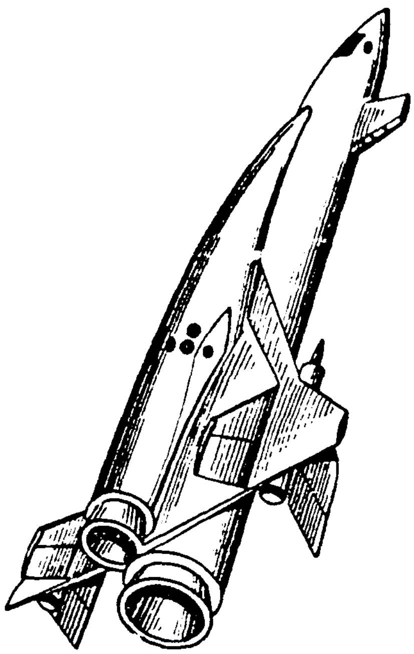

Fig. 82. General view of a rocket booster aircraft.

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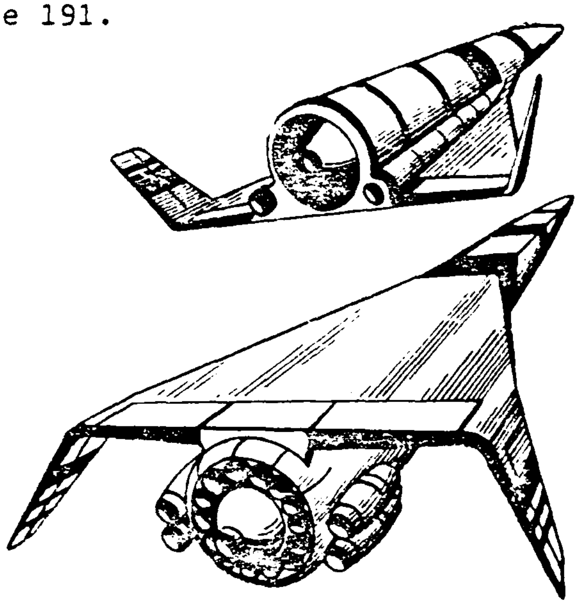
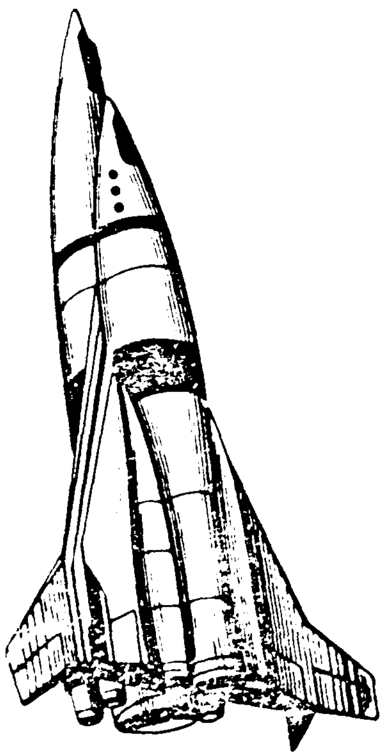


b



a

Fig. 83. Booster aircraft for injection of loads into low geocentric orbit (figure): a) moment of launch; b) after stage separation.

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In both stages there are crews of two people. In the first stage there is installed three ZhRD; the second stage is equipped with one ZhRD. The propellant for the engines is liquid hydrogen and oxygen. Total thrust of engines is 200 t.

For protection of aircraft from heating (to 1500°C) upon atmospheric entry forward sections of stages have ablation coating. Control of aircraft in the atmosphere is produced with the aid of the vertical surfaces, available at the wing tips.

AEROSPACE VEHICLES.

Aerospace vehicles are intended for the delivery of people and loads to orbital stations and back, also, for accomplishing a number of other tasks. Their designs can be very diverse. But the ability to maneuver in the atmosphere with descent from the orbit with the aid of the aerodynamic forces is common for them.

Winged orbital apparatuses (second stages) are shown in Fig. 77-84. But aircraft of this type can and not have a wing. Then their form is selected by such that to obtain the certain lift for the maneuvering in the atmosphere IS of the completion of usual landing. Sucn apparatuses is conventionally designated as apparatuses with the lifting body.

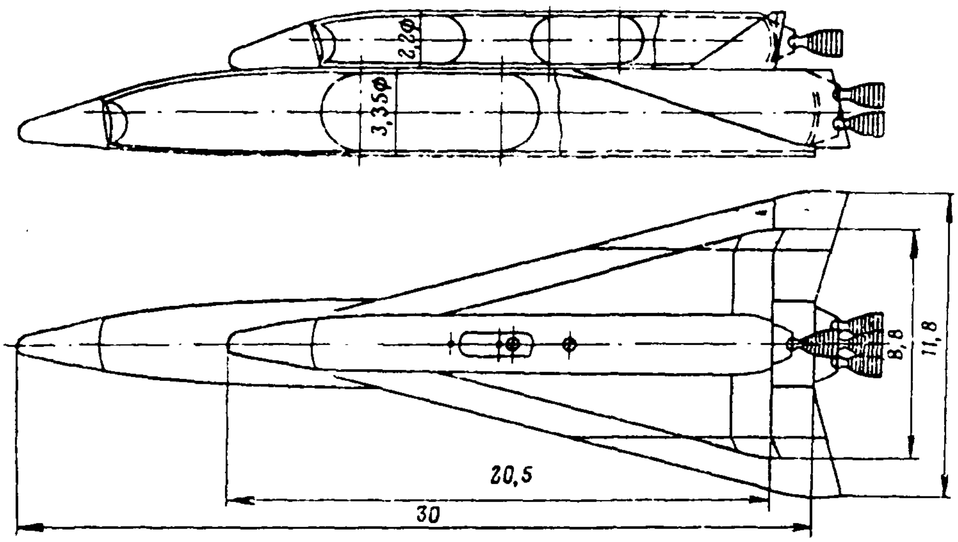

Fig. 84. Schematic of two-stage booster aircraft for delivery of loads to space stations, which are found in orbit at an altitude of 300 km.

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In all the developed diagrams of orbital vehicles designers try to ensure, in the first place, standard conditions for atmospheric entry, especially from point of view of decrease of g-forces, which affect the vehicle and its crew, and decrease of heating elements of construction/design, and in the second place, sufficiently satisfactory maneuverable and landing data.

As far as g-force is concerned, it very sharply decreases in vehicles with a lifting body. They consider that the vehicle with zero lift-drag ratio upon entry into the atmosphere undergoes g-forces equal to 8-10, and with a lift-drag ratio 0.5 they are reduced 4-5 times, i.e., they become equal to 2.

A major advantage of vehicles with a lifting body is the comparatively low heating upon entry into the atmosphere. But their lift-drag ratio even with the very favorable conditions is considerably less than in winged vehicle. At the same time considerably smaller thermal loads affect at the hypersonic speed of flight to the vehicle with the lifting body. This is explained by the fact that in form the given vehicles approach the sphere, which, as is known, has the minimum ratio of surface area to the space. The decrease of this relation is the most effective means of the decrease of the aerodynamic heating of construction/design.

In assignments for development of aerospace vehicles with a

lifting body there was set the goal to achieve upon entry into the atmosphere at hypersonic speeds a lift-drag ratio of from 1.0 to 1.5. This value of quality provides the creation of the vehicle, which combines the advantages of aircraft and rocket. Apparatus makes it possible to complete the satisfactorily planning landing. Its lateral maneuver with lift-drag ratio 1.2 will compose, according to calculations, approximately 1600 km. Furthermore, it makes it possible to ensure the necessary heat shielding. The thermal loads considerably grow/rise with a lift-drag ratio of more than 1.5, which leads to the need for strengthening the heat shield, and consequently, to increase the weight of the vehicle.

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For vehicles with small lift-drag ratio (to 1.0) diagram with lifting body in the form of semicone was selected (angle of semicone approximately 30°). This vehicle provides the best stability of motion. Forebody is blunted for guaranteeing the ablation heat shielding (by ablation of a heatproof layer).

How can there be delivered in orbit vehicles with a lifting body? Three methods can be used for this. The first method of delivery - with the aid of its engines. The second method - the vehicle is first accelerated with first stage of booster aircraft. After uncoupling with it it with the aid of its engines is accelerated to the orbital speed. Finally, for the purpose indicated carrier rockets can be used.

Fig. 85 shows some orbital vehicles with a lifting body. As is evident, they have a different layout and layout.

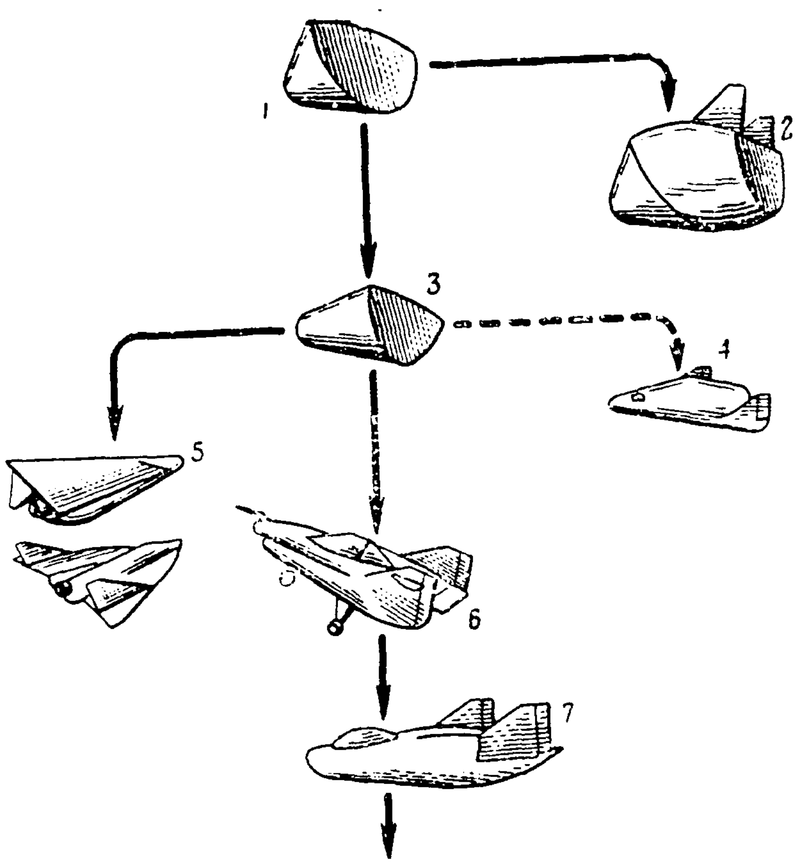

Fig. 85. Flight vehicles with a lifting body: 1 - form of lifting body of vehicle M1; 2 - vehicle M1L; 3 - form of lifting body of vehicle M2; 4 - vehicle HL-10; 5 - vehicle M3; 6 - vehicle M2-F1; 7 - vehicle M2-F2.

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Numeral 1 denotes the American vehicle M1 (with angle of semicone of 30°). At the hypersonic speed it has lift-drag ratio 0.5. The vehicle, which has this quality, is capable to complete lateral maneuver on the order of 400 km. Comparatively insignificant thermal loads upon the atmospheric entry are its advantage. However, it

appeared that a similar aerodynamic shape leads to a deterioration in the characteristics at the subsonic flight speed. In order this to avoid, was proposed the version of vehicle M1 with the inflatable tail section, produced after entry into the atmosphere and decrease in the velocity of subsonic (vehicle M1L). Inflatable tail section increases the lift of vehicle. This provides control capability of it at the subsonic flight speed and satisfactory landing data.

One of such vehicles, created in the USA in manned version, must have a launching weight of a little more than 2000 kgf. During the development of the tail section the length of the vehicle increases approximately doubly, and width by 15%. The lower surface of tail section has a form of the paraboloid of revolution, coaxial with the cone of primary construction. For control of vehicle are elevons and vertical stabilizers. Control surfaces are also from behind, from above and along the sides of the inflatable part of the vehicle. After flight in the hypersonic, supersonic and even transonic modes (upon entry into the atmosphere) the control surfaces of vehicle diverge to the dive. In this case the drogue chute is produced and the fairings about the system of heat shielding together with the fundamental control surfaces are thrown off. Landing equipment and inflatable tail section are produced at the velocity, which corresponds to number M=0.7, at the altitude of approximately 15000 m. Possible sequence of operations during the development/scanning of the tail section of the vehicle is shown in Fig. 86.

For purposes of obtaining higher values of lift-drag ratio at hypersonic speeds of flight for an increase in the range of lateral maneuver angle of semicone of lifting body of vehicle was reduced from 30 to 13°. As a result it was possible to achieve lift-drag ratio 1.2.

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A vehicle with such forms obtained the designation M2. Its lift-drag ratio at the subsonic flight speed about 3. The vehicle has two fins, flat upper parts of the housing and lamps of cockpit projecting above it. Rudders are fastened to the fins. On the aft body there are installed flaps, utilized as the ailerons, elevators and surfaces of trimming. Rudders can be utilized as air brakes for the investigation of their effect on the stability of vehicle. Length of vehicle about 6 m. Launching weight of it is 2700-4100 kgf. It is counted, and this, apparently, it is correct, which the lift-drag ratio of equal to 3, at the subsonic flight speeds is sufficient for obtaining the satisfactory landing data.

According to data of foreign press, there is developed the following procedure of landing vehicles M2: descent in the initial stage with a high constant vertical velocity, alignment of vehicle with constant g- force on the reaching of specific altitude and velocity, low-altitude flight and landing.

Modification of vehicle M2 is vehicle with shaped rear portion.

It obtained designation M2-F1. On it are established/installed the elevons and control surfaces for the path and pitch control. At the end of the fuselage a small rocket engine with a thrust of several ten kilograms is placed. Duration of its operation 10 s.

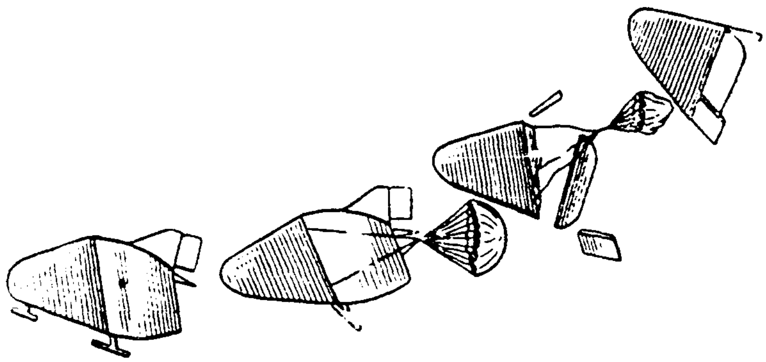

Fig. 86. Possible sequence of operations during development of the tail section of vehicle M1L.

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It is utilized as the emergency system (with inclusion its lift-drag ratio increases from 3 to 8).

For studying the characteristics of this modification vehicle full size with skin/sheathing from plywood was constructed. Its total weight with the pilot and the ejection seat is equal to 500 kgf. Testings was conducted for several months, during which, according to the data of the foreign press, took place about 500 successful flights. In the process of flight tests it was established that vehicle M2-F1 has insufficient stored up lateral stability and view from the cabin, and also that the rudder-effectiveness derivative, arranged/located along the entire trailing edge of fin, is too great.

For eliminating these shortcomings vehicle was modified. Tail section was somewhat elongated, flaps on the lower surface for pitch

control were added. It was called name M2-F2 (it was designated by numeral 7 in Fig. 85). Weight of its 2270 kgf, and with the filled with water ballast tanks - 4080 kgf. Landing speed is equal to 280 km/h.

Variety of vehicles of type M2 is orbital vehicle HL-10 (it is designated by numeral of 4 Fig. 85). It has flatter/plane bottom and more convex upper surface, which makes it possible to achieve lift-drag ratio at the hypersonic speeds more than 1.0 and at subsonic speeds more than 4.0. Vehicle has split rudder and two ailerons with the blunt trailing edges on the aft body. Two supplementary surfaces on each external fin and flaps on the upper part of each aileron raise the stability of vehicle at the transonic and supersonic flight speeds. Control of these supplementary surfaces is accomplished/realized with the aid of the electric motors. Both halves of control diverge to one side. Launching weight of vehicle 2700-4100 kgf, length about 6 m.

As it is indicated in foreign press, by May 1967 on vehicle M2 there were accomplished 14 manned space flights, and on vehicle HL-10 - one manned space flight. Vehicles were hung under the wing of carrier B-52 and were delivered to the altitude of 13700 m, where they were separated/liberated from the aircraft and were lowered under the conditions of gliding/planning.

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Maximum speed in this case corresponded to number M=0.75, landing speed was landing 315-390 km/h, duration of flight - about 4 min.

The carried out flight tests showed that vehicles with a lifting body can safely make a landing. They have a good longitudinal stability (what cannot be said about transverse and directional stability). It was noted that vehicle M2 during landing on 10 May 1967 g, overturned several times, but pilot remained living.

Soon it is proposed to begin the second phase of the flight test program - investigation of flight transonic characteristics. The liquid propellant rocket engine with the thrust 3630 kgf must be established/installed for this on each vehicle.

Subsequently is planned to develop vehicle with lifting body, suitable for flight tests upon atmospheric entry. It is indicated also that is provided for the launching with the aid of the rocket "Titan" of the improved one-place vehicle M2 to a flight trajectory close to the orbital, with an extent of about 90% of a revolution around the Earth. This vehicle must have the maximum similarity to the existing vehicles, which have lifting body. Provision is made for also the development of a multiplace vehicle of the type M2 for the experimental investigations.

In the opinion of foreign specialists, putting into commission of vehicles with high lift-drag ratio can be realized not earlier than

1980th years.

Vehicle M3 (designated by numeral 5 in Fig. 85) in principle differs from all the diagrams examined. The fundamental idea of the schematic of this vehicle consists in reaching/achievement of lift-drag ratio at the hypersonic speeds of more than 1.0 and at subsonic speeds about 10. Structurally this is proposed to solve by changing the geometry of vehicle in the process of flight. Upon the atmospheric entry at hypersonic speeds the wing of vehicle M3 is in the folded state. They consider that in this case it is possible to ensure the sufficiently satisfactory heat shielding of the vehicle, which is a semi-conical body. At subsonic speeds the wing is turned up (Fig. 87), in this case the lift-drag ratio of vehicle increases to 10.

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Very interesting and promising is the manned vehicle with large lift-drag ratio is vehicle SV-5 (Fig. 88) [1].

FOOTNOTE[1]. Vehicle SV-5 is constructed according to two programs: pilotless - SV-5D and piloted - SV-5P and SV-5J. The turbojet engine is installed on the latter. ENDFOOTNOTE.

It has well streamlined wedge shape. Small vertical fins and flaps in its rear part provide supplementary maneuverability. The weight of vehicle 4500 kgf, length - is 7.3 m, and spread of 4 m.

For the creation of the vehicle SV-5, there was constructed its model with a weight of approximately 400 kgf. The model is coated by the heat shield, which has sandwich construction and flat external surface, which does not melt, but it is charred, providing the retention of aerodynamic shape and the rigidity of construction/design.

First launching of the model was realized during December of 1966 with the aid of rocket "Atlas". The vehicle was ejected to the base altitude, then it entered into the dense layers of the atmosphere along a trajectory close to the calculated, and then - into the landing region.

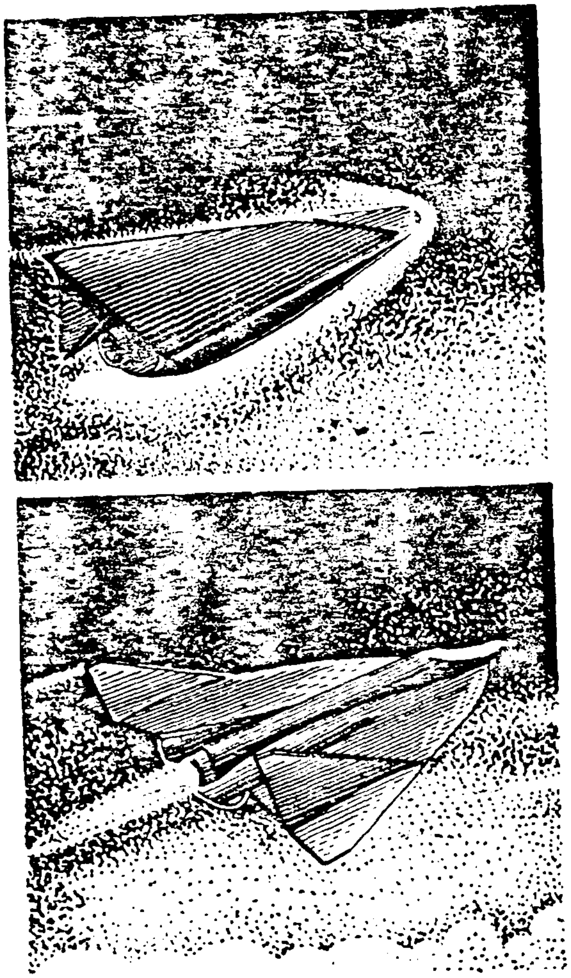

Fig. 87. Vehicle M3 with folded (above) and expanded/scanned (below) wing.

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The position control of vehicle and of flight speed beyond the limits of the atmosphere was accomplished/realized with the aid of the jet nozzles working on nitrogen, and in the atmosphere - flaps. In the lower layers of the atmosphere the vehicle had subsonic flight speed; the guidance and control system, instruments and electrical support

systems functioned normally. During entire flight the vehicle could maneuver and it was aerodynamically stable.

During March 1967 with the aid of the rocket "Atlas" there was realized a second launching of the model. As the foreign press reports, the model made a successful flight, and it completed splashdown. It was impossible to save it, and it sunk.

On one of the stages of the study of vehicle SV-5D, there is provided for its jettisoning from a B-52 aircraft. In this case it must be equipped with the rocket engine, with the aid of which it is proposed to investigate maneuvering vehicle in the range of velocities from 2100 km/h to the rate of landing.

According to reports of the foreign press, one of last experimental vehicles with a lifting body, constructed in 1967, is the vehicle "Martin X-24A". He is intended for conducting flight tests according to the program "pilot" (testing the manned vehicles with a lifting body at low speeds of flight) and is the streamlined body of triangular planform with the convex upper and flatter/plane lower surface.

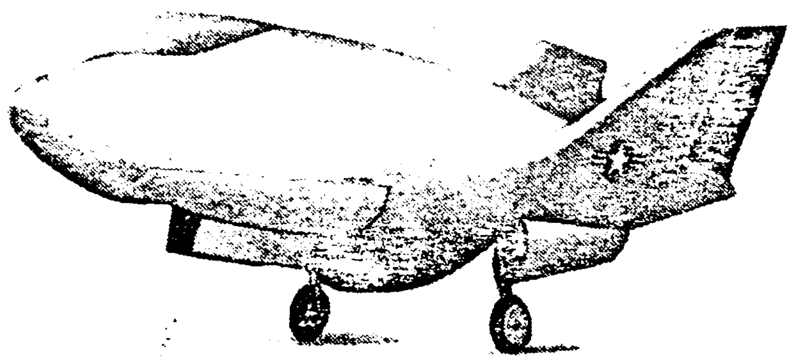

Fig. 88. Vehicle SV-5J.

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The area of the carrier surface is about 18 m^2. Vehicle has the ejectable cockpit hood of pilot and the ejection seat, which ensures abandoning aircraft at any flight speed and at any altitude. The landing gear is retracting, with a two-wheeled front and single-wheeled main strut.

Power plant of the vehicle consists of one ZhRD with a thrust of about 4000 kgf and two ZhRD with a thrust of 230 kgf, which operate on hydrogen peroxide and used during landing. Launching weight of the aircraft was about 5000 kgf. The maximum speed of flight at the altitude of 30000 m must correspond to number M=2.

Lift-drag ratio of vehicle at subsonic speeds 4-6, on hypersonic - 1.1-1.4.

On vehicle there are aerodynamic control surfaces - two upper and two lower flaps and split rudders on two external vertical fins. The upper and lower flaps, which work as the elevons, are utilized for the longitudinal and lateral control, upper rudders - for the lateral guidance, and lower - for the trimming of the vehicle.

Vehicle has feed system of air into cabin/compartment and aggregates/units of heating.

First tests of vehicle "Martin X-24A" is intended to carry out without engine. The vehicle will be hung under the wing of the bomber B-52 and be separated from it at an altitude of 12000-15000 m at speeds of more than 800 km/h, and then there will be complete gliding descent and landing.

During subsequent starting/launching vehicle will be equipped with rocket engine. After the department/separation of vehicle from aircraft B-52 this engine will derive it on the maximum altitude of 30000 m and will report the maximum speed, which corresponds to number M=2, with which the vehicle will enter into the dense layers of the atmosphere. The landing speed of the vehicle is 260-360 km/h. Duration of flight from the moment of separation from the B-52 aircraft to landing is 15 min.

On the basis of results of investigations, obtained with the aid

of this vehicle, they intend to create a space vehicle intended for servicing the orbital station.

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Above it is evident from that presented that in airframe structures with a lifting body there is accepted a compromise solution: on the one hand, there are provided sufficiently satisfactory inlet characteristics into dense layers of the atmosphere, and on the other - satisfactory landing data.

They, nevertheless, consider that to most fully satisfy both those and other requirements is possible only by means of applying two-stage designs of orbital vehicles.

As an example of this design there can serve the American orbital vehicle "Janus" (Fig. 89). Its first stage is the housing of the blunted conical shape with an apex angle of 24°, which creates lift. This housing is similar to the vehicle 1 shown in Fig. 85. Delta-wing airplane, which is the second stage, is installed on top of it. Fins at the wing tips are placed in the unsealed deepenings in the housing of vehicle. The cockpit canopy protruding above the upper wing surface provides a coverage in flight in space and the atmosphere. The fuselage of the aircraft with the turbojet engine located in it is completely drowned in the construction/design of vehicle.

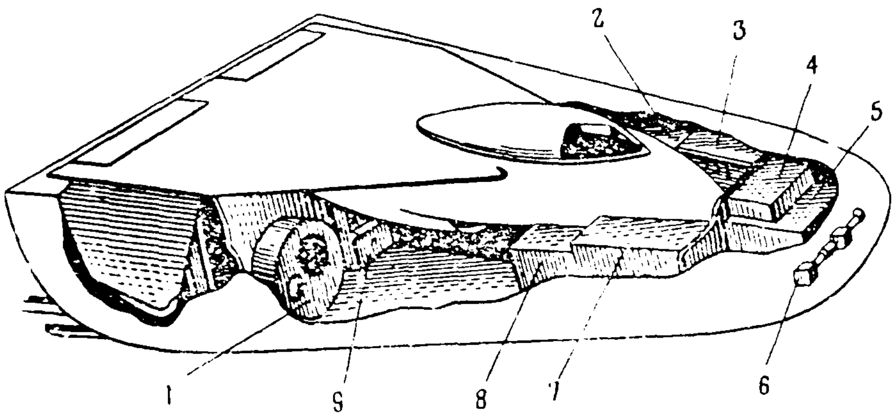

Fig. 89. Orbital vehicle "Janus": 1 - airlock; 2 - food products; 3 - panel of life-support system; 4 - connected radio equipment; 5 - heat shield; 6 - system of control and navigation; 7 - control system; 8 - vessels for positioning water and the toilet; 9 - instrument panel.

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Form of vehicle (semicone with flat upper surface) provides its easy mating with the aircraft. The upper surface of the delta wing of aircraft forms the part of the upper surface of vehicle. This makes it possible to keep its aerodynamic shapes constant.

All internal sections of the space vehicle between the front and rear sealed baffles, including the fuselage of aircraft, are hermetically sealed. Electronic equipment, systems of power supply and life support, heat exchangers and another fixed equipment are placed in the forward section of the vehicle. In its middle and rear parts the crew compartments and sections with the special equipment

are located. In the rear end of the vehicle after the sealed partition is an airlock, utilized for the entry into the space vehicle and the output from it on the Earth and in space. In the unsealed tail section the rocket engines of stabilization and control system of the position of vehicle in the space and brake rocket engine are established/installed. For trimming of vehicle upon entry into the atmosphere are four controlled flaps, fastened to the hinges along the periphery of the tail section of the vehicle. The system of balancing/trimming allows for the possibility of changing the lift-drag ratio of vehicle.

Fundamental controls and instruments are established/installed on the control panel in vehicle, they are necessary for accomplishing different operations in flight along orbit, including orbit ejection and orientation for entry into the atmosphere.

Aircrew consists of three people. In flight they can freely pass from the compartment of aircraft into the vehicle and back.

Flight program under normal conditions consists of starting/launching, orbital flight, entry into the atmosphere, descent and landing. Vehicle in orbit is started with the aid of the booster aircraft or the carrier rocket.

As it is indicated in foreign press, this vehicle, designed for two-day flight along low orbit, must weigh approximately 7500 kgf.

Lift-drag ratio of vehicle can vary within the range of 0.35 on 0.75 in flight at first stage and reach approximately 10 in flight at second stage.

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This makes it possible to increase flying range in the stage of entry into the atmosphere on 2200 km or to decrease on 1500 km relative to the calculation landing spot. According to the calculations, lift-drag ratio must be sufficient for the accomplishment of the large-scale lateral maneuver, which ensures the selection of necessary landing place.

Vehicle, as they assume, will be aerodynamically stable upon entry into the atmosphere and in all other stages of flight at subsonic, transonic and hypersonic speeds.

The housing of the vehicle has double walls. The internal wall of its sealed part is made from sheet aluminum, and the outer covering - from the laminar aluminum panels with the honeycomb filling. The external panel of the outer covering is closed with heat shield from the ablating material. The space between the external and internal walls of housing is filled with light fibrous or foam thermal insulation material. As it is indicated in the foreign press, such heat of shield protects aluminum sandwich construction upon its entry in the atmosphere from the heating. The temperature of external wall

in this case does not exceed 315°C, but on internal wall - it is higher than 21°C.

This large temperature differential between external and internal walls with their rigid connection would lead to considerable thermal stresses in the framework and sandwich constructions. Therefore, for fastening of the outer covering to the framework/body flexible couplings are developed.

Temperature of heat shield and internal sections of vehicle during orbital flight is regulated by circulation of liquid along manifold, built in fundamental elements of construction/design of vehicle. For maintaining the prescribed/assigned temperature of internal sections the rear wall of vehicle have passive emitters with the coatings, which possess very different absorption characteristics and emission of heat.

System of power supply consists of fuel cells. The same fuel cells provide crew with water and through the heat exchanger are connected with the control system of temperature.

Life-support system regulates temperature and air pressure in cabin/compartment.

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Air composition is maintained similarly: 47% of oxygen and 53% of

nitrogen. Special filter removes carbon dioxide. Water in this case is separated/liberated by freezing.

Vehicle has stabilization and control system of its position in space. This system works with the aid of two solid-propellant jet engines, established/installed at angle of 25° to the axis/axle of vehicle.

It is assumed that communication system will consist of equipment for voice communication and transfer of telemetering data during the launching, orbital flight and upon entry into the atmosphere. Centimeter wave band for decreasing the effect, connected with the effect of ionization, will be utilized upon entry into the atmosphere.

Under normal operating conditions, and manual, the automatic or combined control of vehicle also in emergency situation is provided for. In flight along the orbit the position control of vehicle in the space is accomplished/realized with the work of engines under the conditions of low thrust.

Orbit ejection for landing is accomplished/realized by firing reverse jato. In this case by the gyrostabilized platform, error signals are issued, and the engines are switche to the mode of full thrust.

Under normal conditions the pilot accomplishes the orbit

ejection. Before firing of brake motor the vehicle must be properly oriented. The data, necessary for orienting of vehicle and precise selection of the time of firing brake motor, provides the indicator, which continuously indicates possible landing places in the dependence on the moment of inclusion and direction of thrust of the engine. After the switching on of brake motor the vehicle for 14 min descends to the dense layers of the atmosphere. Onboard calculator at this time determines the value of lift-drag ratio, necessary for the landing in the prescribed/assigned area. The flaps, which ensure the selected value of lift-drag ratio, diverge after this, and vehicle is oriented for the gliding flight.

As they assume, the angle of slope of initial trajectory of entry into the atmosphere must be in initial section approximately 2°.

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The vehicle, having lift, continues to fly at the high altitude, and the stage of braking "is extended". This provides reduction in the maximum g-forces to 2 instead of 8, which can be expected under the same conditions in the vehicles which do not develop lift.

Flight path angle sharply increases 700-800 s after orbit ejection and entry into the atmosphere virtually concludes. In the finite segment of reduction/descent the vehicle flies with transonic speed at the altitude of 15000 m.

At altitude of 9000 m, at flight speed, which corresponds to number M=0.6, aircraft is separated/liberated from vehicle and is continued flight, to landing place with the aid of a turbojet engine[1].

FOOTNOTE[1]. The altitude of 9000 m is calculated. It can be increased to 15000 m. ENDFOOTNOTE.

Creation of trouble-free system of department/separation is considered one of most difficult problems during development of this vehicle. One of the systems of department/separation must consist of three pyromechanisms - one of the nose section and two at the wing tips on the rear longeron/spar. The TRD of the aircraft is planned to include at altitudes of not more than 14000 m and the velocity, which corresponds to number M=0.9. It is assumed that the flight and landing data of aircraft will be compared with similar characteristics of contemporary jet aircraft. The landing speed of aircraft is provided for about 200 km/h.

Creators of vehicle consider that on its qualities it can provide possibility of its use for exploration, inspections with servicing satellites, and also for transport transportation, assembly or repairing space stations in orbit.

It should be noted that extensive scientific research work on study and resolution of different problems of flight into space at present is conducted.

For the development and substantiation of requirements for a system of controlling a vehicle, which possess lift, widely studied was the entry into the atmosphere of flight vehicles, in particular, the aircraft X-15 (Fig. 90), although, as American specialists note, this aircraft was created not for the study of entry into the atmosphere of orbital flight vehicles.

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According to reports to the American press, there were accomplished sixteen test flights of aircraft X-15 to high altitude and were mastered modes of entry into the atmosphere and of flight with low velocity heads requiring for its return. In these flights ascended vertical distance to 108 km at the apogeal velocity of trajectory about 1500 m/s. Angle of attack upon entry into the atmosphere reached 26°, the normal load factors exceeded 5.5, and maximum value of velocity head reached 22500 kgf/m^2.

In connection with the fact that velocity of aircraft X-15 is considerably lower than velocity of orbital vehicles, time of its entry into the atmosphere is less than time of their entry (Fig. 91).

Aerodynamic guidance of aircraft was provided by vertical tail assembly (azimuth guidance) and horizontal tail assembly (longitudinal and lateral control).

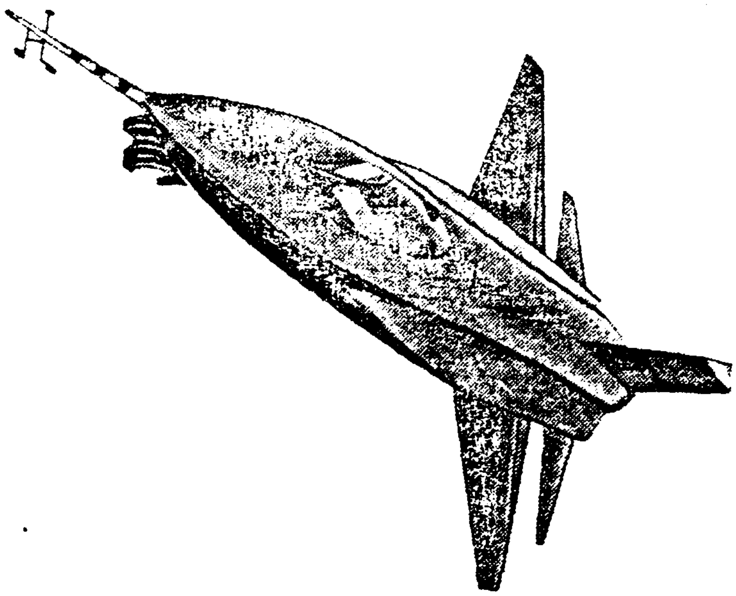

Fig. 90. Aircraft X-15.

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Control surfaces were actuated by hydraulic boosters. The forces of control were created by spring automatic machines. For countering of the angular velocity of aircraft the system of an increase in the stability provided supplementary aerodynamic damping by the deviation of aerodynamic control surfaces.

Each aircraft was equipped with jet nozzles for control with low velocity heads. Jet guidance of aircraft was intended for maintaining the prescribed/assigned position of aircraft in the space, when the

effectiveness of aerodynamic control surfaces was insufficient.

One of the aircraft was equipped with self-tuning system for control, which works on signals of deflection velocity. This system makes it possible to produce automatic balancing/trimming, limits g-forces by the permissible values, is accomplished/realized the automatic connection/communication of aerodynamic and jet guidance, etc. It gives the possibility to parry the errors of control of pilot, provides the increased reliability and safety in the case of failures. The latter is reached by the fact that the system of jet guidance of aircraft is backed up.

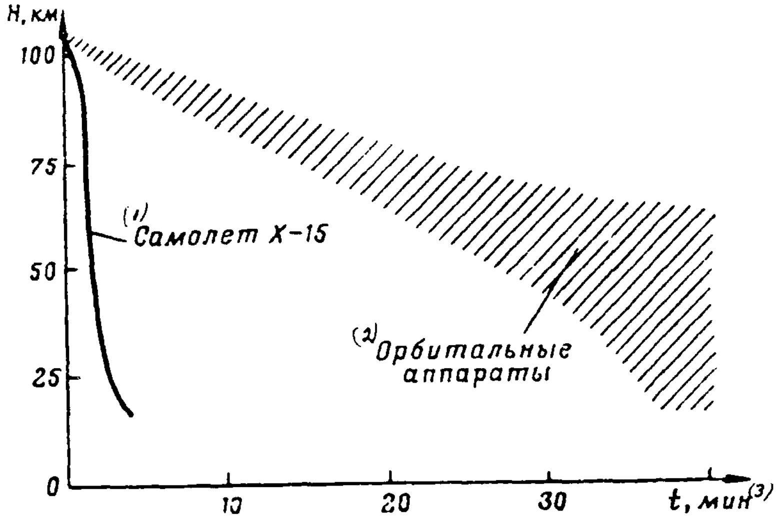

Fig. 91. A comparison of time of entry into the atmosphere of aircraft X-15 and orbital flight vehicles which use lift; lift-drag ratio of 1.0-2.0.

Key: (1). Aircraft X-15. (2). Orbital vehicles. (3). min.

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Entries of the aircraft in the atmosphere with low velocity heads were produced with application of system of jet guidance[1].

FOOTNOTE[1]. The system of reactive control was utilized upon the entry to considerably larger velocity heads than it was assumed. ENDFOOTNOTE.

In this case the fuel consumption in control system and some special features of its use were determined. In the press/printing it is indicated that first stored up fuel/propellant, requiring for the

stabilization of aircraft and maintaining of the specific angle of attack upon entry into the atmosphere, it was determined during the flight simulation of aircraft X-15 at rated altitude of 75 km. However, in the first altitude flights it became clear that more fuel/propellant is required how it was assumed. Therefore there were Illegible the system of pumping fuel/propellant from the primary fuel-1 system of aircraft, the engeine of reactive control ensured the feed/supply.

It is interesting that pilots observed loss of stability of aircraft with respect to velocity, in particular, during final approach, when it is necessary to concentrate attention on a surrounding situation. Pilots noted that the limiter of the normal load factors established/installed on the aircraft raises safety, since the g-forces, which appear upon entry into the atmosphere, approached maximum for the construction of the aircraft. They noted also that because of the application of the self-tuning system for control entire flight, with exception strictly of landing, can be made automatically.

It should be noted that self-tuning system for control proved to be sufficiently reliable. Only the one component of system failed in two years of tests. However, this did not disable it. In this case the track angle only somewhat changed, which the pilot defined as a considerable path imbalance.

Before each flight all operations of pilot on entry into the atmosphere of aircraft X-15 were mastered first on trainer. The mechanics of control in this case corresponded to flight conditions.

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For safe return of aircraft of space not less important than control of its position for purposes of stabilization, is control power dissipation rating or of flying range, although control of flying range is not for aircraft X-15 so serious a problem as for orbital vehicles.

In flights to maximum altitudes maximum range from start to landing approached 500 km.

Within small time of entry into the atmosphere change in lift-drag ratio little affected flying range. But after entry into the atmosphere its change significantly affected flying range. It is possible even to say that, utilizing it, pilot established the necessary line of descent and brought aircraft into the area of landing place at the altitude of 6 km at the flight speed, which corresponds to number M=0.8. Landing was produced on the previously selected area.

By fundamental elements of control flying range were angle of attack and air brakes. In flight at the angle of attack, which corresponds to maximum lift-drag ratio, the pilot reached maximum

range. By the application of air brakes and by a change in the direction of flight was provided minimum flying range.

Important result of flights of aircraft X-15 was study of aerodynamic heating. First were designed the local flows of heat in connection with the conditions, expected in flight. The obtained results were utilized for calculating the heat transfer to the aircraft. Then the temperature of the sheathing was determined and heating inner zones of construction/design as a result of thermal conductivity or emission was designed. Calculation data and temperatures of wing actually measured in flight at an altitude of 96 km coincide sufficiently accurately. However, although heating different sections of aircraft was designed previously, during the flights there were the cases of damaging some sections, since actual temperature exceeded calculated. Thus, for instance, in one of the flights, when there was reached the maximum speed of 7300 km/h at an altitude of 30 km, burned-out ablation coating on air brakes, and on the leading wing edge it turned black.

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Chapter 7.

MATERIALS USED IN STRUCTURAL ELEMENTS OF SPACE VEHICLES.

The fundamental factor which has effect on strength of structural elements of hypersonic aircraft is aerodynamic heating (Fig. 92). Maximum temperature is on the nose fairing and leading edges of the lifting surfaces of wing and fuselage. Furthermore, high temperatures appear in the clearances between the fixed and moving surfaces, at the points of the constructions/designs, with which are encountered the shock waves, and also in the places located near the sections of interaction of jumps and, etc.

So that construction/design could work in severe temperature conditions, are used special high-melting materials, which can maintain/withstand aerodynamic heating without deterioration in strength data.

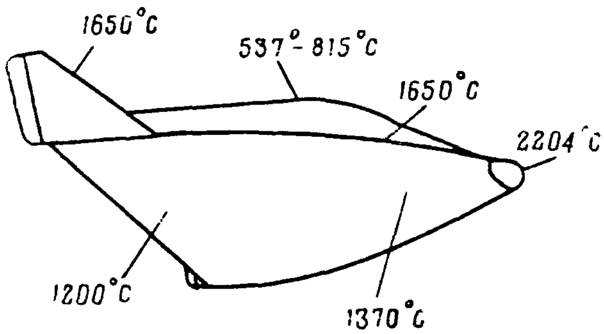

Fig. 92. Heating of the construction of hypersonic aircraft.

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Whole structural elements can be prepared from these materials. They can be utilized also as protective coatings. Besides these materials, in the construction/design of hypersonic aircraft there are utilized materials with the high heat-radiating and heat-absorbing properties, which ensure the effective heat removal from the most heated parts of the aircraft, and the active methods of cooling the elements of construction/design also are used.

Such special materials, as it is noted in the foreign press, are the metals - titanium, molybdenum, niobium, tungsten, tantalum and alloys on their basis, nonmetals - graphite, ceramics, etc.

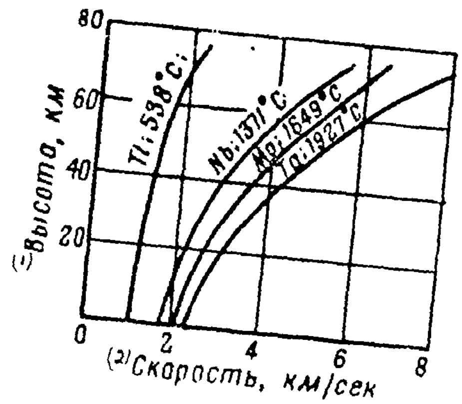

Fig. 93. Limiting values of temperatures, which can withstand high-melting materials.

Key: (1). Altitude, km. (2). Velocity, km/s.

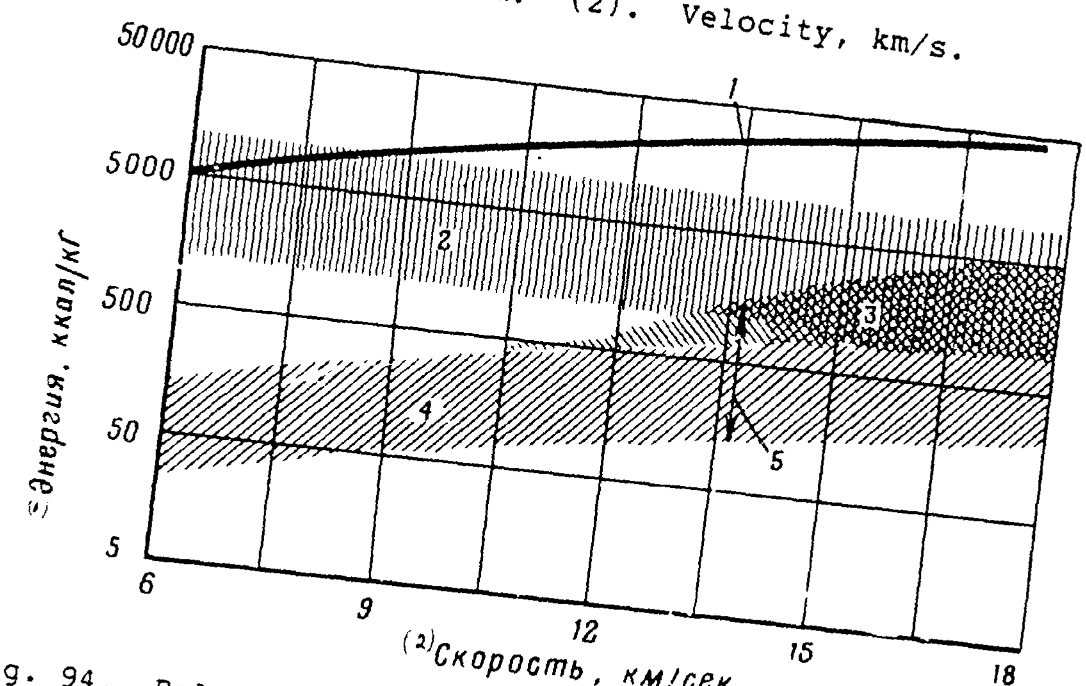

Fig. 94. Role of heat shield and different methods of scattering: 1 - kinetic energy of space vehicle; 2 - heat shield; 3 - emission; 4 - convection; 5 - thermal load on vehicle.

Key: (1). Energy, kcal/kgf. (2). Velocity, km/s.

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Temperatures which can withstand some of the metals indicated are shown in Fig. 93. They are reached on the lower wing surface at a distance of approximately 1.5 m from the leading edge.

For less heated elements of construction of aircraft will be as before widely utilized existing structural materials.

Fig. 94 shows role of heat shield and different methods of energy dissipation in reduction in thermal load of space vehicle. The thermal load, which affects on the vehicle, it is shown as the velocity function of the entry of vehicle in the atmosphere. As is evident, basic part of the heat is scattered in the atmosphere and its only small quantity in the form of thermal load falls on the surface of vehicle. At the low speeds the heat is transmitted to vehicle mainly as a result of convection. The role of heat transfer as a result of the strongly heated shock wave considerably grows/rises at high velocities.

So that elements of the vehicle could operate in the assigned temperature conditions, there can be used constructions/designs of heat-absorbing and heat-radiating types. The heat-absorbing constructions/designs make it possible to absorb a large quantity of heat or to abstract/remove heat by cooling "exudation" and by the escape of material (ablation). The heat-radiating constructions/designs are subdivided into the "cold" (cooled) and "the

hot". In the heat-radiating "cold" constructions/designs fundamental elements are thermal protective shields. Because of this protection the elements of load-carrying structure have comparatively low temperatures and usual materials are applicable for them. One of the types of the construction/design of heat shield is shown in Fig. 95. A fundamental requirement, which is presented to the heat shields, is the following: heat shields must have low thermal conductivity and low weight.

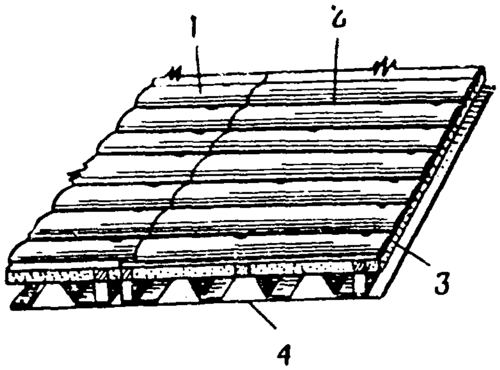

Fig. 95. Heat shield: 1 - slightly fluted metallic sheet; 2 - bracing strut; 3 - insulation which withstands high temperatures; 4 - primary construction.

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In "hot" constructions/designs sharply increases the heat dispersion into the environment by its emission.

It should be noted that materials with low values of the coefficient of thermal conductivity have a very low specific strength, which comprises portions of specific strength of aluminum alloy. This is, of course, their great shortcoming. But it "is overcome" by the fact that only for some elements of construction/design the determining requirement is the requirement of the presence of a sufficient tensile strength. However, the voltage/stress of loss of stability and the modulus of elasticity is most important for the majority of the elements. But these values strongly are not reduced with an increase in the temperature.

Fig. 96 shows standard patterns of the design of thermal insulation.

Elements of hypersonic aircraft, which can be subjected to action of temperatures, which exceed maximum permissible for high-temperature (strength) metals (~1700°C), are intended to manufacture from graphite, high- melting ceramics, cermets or to cool them.

Fig. 96. Standard patterns of construction/design of thermal insulation.

Key: (1). Metal shell. (2). High-temperature insulation/isolation. (3). Holding rod. (4). Laminar sealing. (5). Channels for purging. (6). Heat-radiating sheets and separator. (7). Insulation with purging. (8). Insulation with sealing. (9). Construction/design with evacuation. (10). Sealing. (11). Radiation barriers.

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In particular, the air intakes of the engines, combustion chamber wall and nozzle it is proposed to satisfy two-wall and three-wall construction/design with the channels for the coolant. As the coolant they intend to use liquid hydrogen, which is fuel/propellant for the engines.

External wall of housing of hypersonic aircraft will be manufactured from high-temperature (strength) metals with protective coating from niobium (where it works when $t>1000$ °C) and tantalum (where it it works when $t>1480$°C). At lower temperatures it is proposed to utilize a usual high-temperature (strength) alloy. Internal walls will be manufactured from the high-temperature (strength) alloys, and also the titanium and aluminum alloys - depending on the possibilities of the cooling systems.

This is how the construction of wall of housing of one of experimental vehicles is made. The wall with a thickness of about 50 mm consists of the outer covering of special alloy with the thickness of 0.25 mm, layers of thermal insulation with the thickness of 25 mm and honeycomb aluminum constructions/designs by thickness ~25 mm. 1 m² of the wall of the housing of vehicle without the honeycomb construction/design weighs 7.5 kgf, and with the honeycomb construction/design, 22 kgf.

Coatings from high-temperature (strength) metals and their alloys can be used in frameworks at temperatures up to 1600°C. High-melting ceramics, probably, will find use as the unloaded coatings, which work at a temperature of 2100°C. Some properties of these materials are given below.

Molybdenum. Molybdenum and its alloys possess very high high-temperature (oxidation-resistant) properties in the loaded state.

However, temperatures, with which they work, are higher than temperatures of their recrystallization, and in this state the alloys develop a tendency toward brittleness even at room temperature, especially under the effect of impact load. Welds are also inclined to brittleness and are unsuitable for the application in the frameworks.

Niobium. Niobium and its alloys have the melting point (2468°C) lowest among the refractory metals. According to existing knowledges, in niobium, furthermore, the lowest value of the modulus of elasticity. But plasticity and the weldability of niobium good.

Tungsten. This metal has highest melting point (3380°C). Therefore it should be utilized for the constructions/designs, which work in the wide temperature range.

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However, it with difficulty is treated into the sheets being deformed. Welds made of the tungsten do not retain plasticity at room temperature. In the recrystallized state it is brittle at a temperature lower than 200°C. Tungsten density higher than density of other metals in question.

Tantalum. Tantalum possesses the melting point, close to the melting point of tungsten (3000°C). It can be prepared in the form of the sheets, deformed with the application of usual technology and is

capable to retain plasticity in the recrystallized state at room temperature. It is utilized for the production of welds. The modulus of elasticity of tantalum is higher than the modulus of elasticity of niobium, which is also considered suitable for manufacturing the welded constructions. Negative property of tantalum - high density, equal to 16.6 g/cm^3. Moreover it by very expensive and least propagated from the base refractory metals.

Difficulty, connected with application of metals examined and their alloys, consists in the fact that they at high temperatures intensely are oxidized. Therefore, protective coatings are necessary for them. Used as such coatings most frequently are silicides, aluminides, beryllides and oxides. Oxidation at high temperatures occurs with the intensity, which depends on the level of effective stresses. But about this there is very scanty information. Moreover this information is obtained under the varied conditions for tests, which very impedes the comparison of results. However, it is established that in all cases the oxidation occurs with the constant intensity until a certain critical stress level is reached. Then the intensity of oxidation rapidly grows/rises.

New low-temperature process of precipitating film of zirconium dioxide to metal is at present developed. Vapors of zirconium tetrachloride and phosphorous chloracid are transferred by hydrogen in a gaseous state to the surface of the metal, heated to a temperature of 870-920°C. In this case on it a film of zirconium dioxide is

settled. The film possesses a noncrystalline structure, it is solid, it has a good adhesion with the surface, it is transparent and nonconducting.

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Tests for determining its mechanical and thermal characteristics now are conducted. It is assumed that the film of zirconium dioxide can be used for the protection from oxidation at temperatures approximately 3700°C.

High-melting ceramics can be used for manufacturing of intake (leading) edge and lining of combustion chamber of engine. For the entering edge is necessary the ceramics, which could withstand not only high temperature (1600°C), but also considerable pressures (to 2810 kgf/cm^2). The lining of the combustion chamber of engine must be made from the ceramics from properties somewhat different than at the entering edge. Here it is not compulsory so that the ceramics would maintain/withstand high pressure; however, it must transfer sufficiently well the effect of temperature on the order of 2100°C. It is desirable even so that it would be designed for the use at even higher temperatures (to 2800°C).

It should be noted that all ceramic materials, except oxides, have limited oxidation resistance at high temperature. Therefore under such conditions they undergo considerable changes.

High resistivity to the effect of oxidation at high temperatures can be obtained with formation of glassy film of silicon dioxide. Consequently, the materials, which contain its, such, as carbide, nitride of silicon and molybdenum disilicide, from this point of view are of greatest interest. Nitride of silicon does not maintain/withstand high temperatures, but nevertheless it can find use at a temperature about 1600°C.

Ceramics is interesting fact that it easily is molded. However, further methods of its working/treatment are very labor-consuming and road.

On hypersonic and aerospace vehicles as fuel/propellant it is proposed to utilize liquid hydrogen. It has calorific value, three times greater than hydrocarbon fuels. But its specific weight/gravity is 11 times lower than the specific weight of hydrocarbon fuels. Therefore the space of fuel/propellant necessary for achievement of the prescribed/assigned flying range, during the use of liquid hydrogen is much greater than with the hydrocarbon fuel.

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Liquid hydrogen has a boiling point -253°C[1].

FOOTNOTE[1]. Temperatures in literature indicated are called also cryogenic (from the Greek words kryos - cold, ice and genos - birth). ENDFOOTNOTE.

This creates great difficulties during its storage. The fact is that the brittleness of structural materials strongly is raised at this temperature of hydrogen, as a result of which to utilize them for vessels for hydrogen is impossible. Titanium is the most suitable material for storing hydrogen. Its strength at a temperature -253°C approximately is two times more than with the room. It retains also a sufficient ductility and resistance to brittle failure.

Tanks with liquid hydrogen must be well heat-insulated in order to avoid evaporation of hydrogen (Fig. 97).

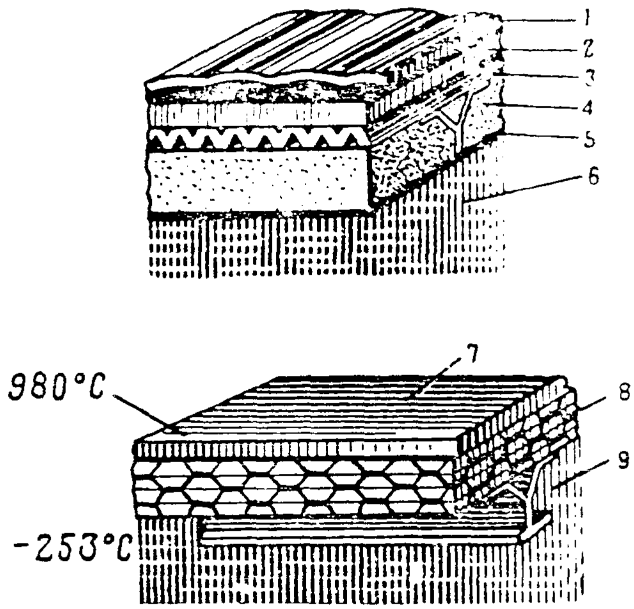

Fig. 97. Types of structures of storage tanks of hydrogen: 1 - shield (980°C); 2 - fibrous insulation/isolation; 3 - primary construction; 4 - cryogenic insulation; 5 - tank; 6 - liquid hydrogen; 7 - shield; 8 - multilayer design; 9 - liquid hydrogen.

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It is possible to note in conclusion that the decade mastery of space (beginning from the launch of the first Soviet Earth satellite) is characterized by outstanding achievements in the development and improvement of space vehicles.

Of course, humanity will not stop at this. The genius of man, who created vehicles for flights through the air and space, will master the limitless spaciousness of the universe.

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